NASA Conference Publication 10116

Nuclear Propulsion Technical Interchange Meeting

Volume I

Proceedings of a meeting held at NASA Lewis Research Center
Plum Brook Station
Sandusky, OH
October 20-23, 1992

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NASA Conference Publication 10116

Nuclear Propulsion Technical Interchange Meeting

Volume I

Proceedings of a meeting sponsored and hosted by
NASA Lewis Research Center
Plum Brook Station
October 20–23, 1992

National Aeronautics and
Space Administration

Office of Management

**Scientific and Technical
Information Program**

1993

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

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PREFACE

Robert R. Corban
Nuclear Propulsion Office
NASA Lewis Research Center

The Nuclear Propulsion Technical Interchange Meeting (NP-TIM-92) was held at NASA Lewis Research Center's Plum Brook Station in Sandusky, Ohio on October 20-23, 1992. Over 200 people attended the meeting from government, Department of Energy's national laboratories, industry, and academia. The meeting was sponsored and hosted by the Nuclear Propulsion Office at the NASA Lewis Research Center. The purpose of the meeting was to review the work performed in fiscal year 1992 in the areas of nuclear thermal and nuclear electric propulsion technology development.

These proceedings are an accumulation of the presentations provided at the meeting along with annotations provided by the authors. All efforts were made to retain the complete content of the presentations but at the same time limit the total number of pages in the proceedings.

I would like to acknowledge the help and support of a number of people that have contributed to the success of the meeting:

(1) Daniel S. Goldin, NASA Administrator, for taking the time to eloquently contribute to the meeting as our keynote banquet speaker,
(2) the Session Chairmen, for organizing excellent technical content for their sessions and keeping the sessions on-time,
(3) the authors, for describing their results and accomplishments,
(4) our host, Robert Kozar and his dedicated staff at the Plum Brook Station, for providing an excellent facility for the meeting and an commendable tour of their world-class test facilities.
(5) and finally to all the "behind-the-scenes" people that were so instrumental in making the technical interchange meeting a success - especially Bonnie Kaltenstein and Jean Roberts, whose excellent organization and orchestration of the meeting was the key to its success.

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

INTRODUCTION

REQUIREMENTS

N93-26909

Mars Exploration Program

EXPLORATION PROGRAMS OFFICE

Office of Exploration

Mars Exploration Program

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 20, 1992

Dwayne Weary
Exploration Programs Office (EXPO)
NASA Johnson Space Flight Center

Space Exploration
Missions to the Moon, Mars, and Beyond ...

America wants a NASA of explorers, pioneers, and innovators to boldly expand the frontiers of air and space for the benefit of all.

Assure America's Leadership in the Next Millennium

Advance Scientific Knowledge

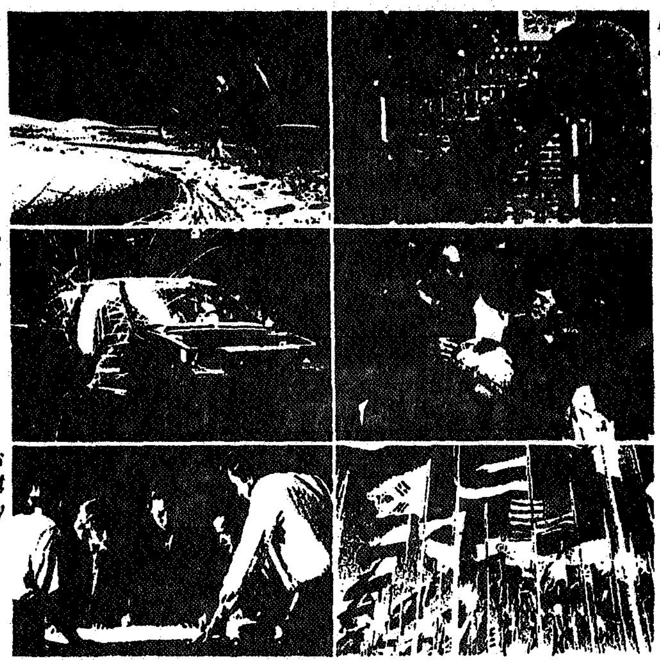
Drive Technology Advances

Boost our economic competitiveness

Enrich Education and Scientific Literacy

Rebuild America's Management Capability

Promote International Cooperation



Mars Exploration Program

- SEI Program Plan -

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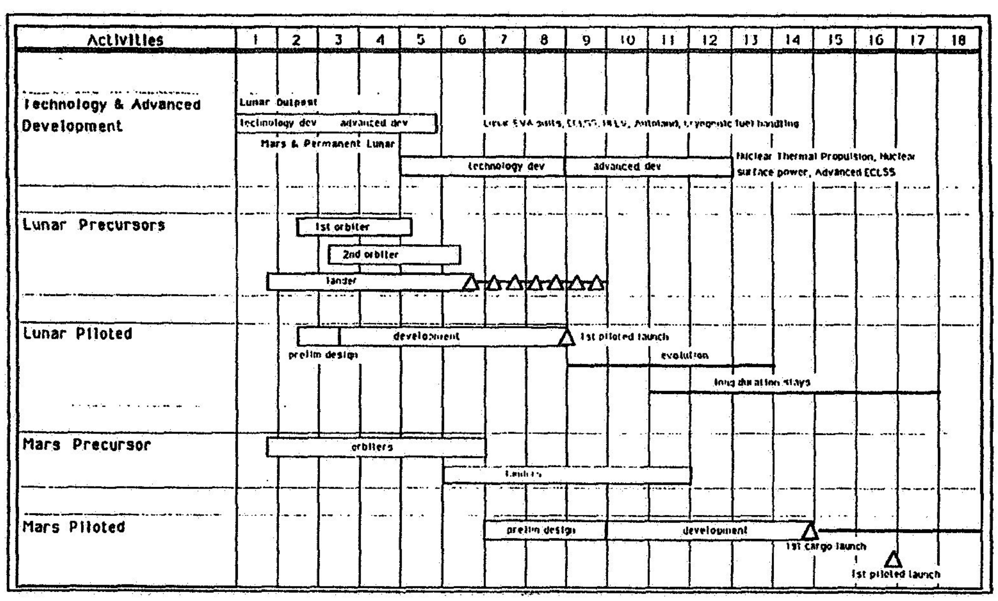

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Mars Exploration Program
- Program Goals -

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Technical Goal

Verify the ability of people to inhabit the planet Mars

Management Goal

Demonstrate effective global cooperation in a high-technology initiative

Societal Goal

Demonstrate improvements in economic vitality and the quality of life for all participating nations

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Mars Exploration Program
- Surface Mission Objectives -

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- ❑ Demonstrate substantial self-sufficiency in life-support consumables and in fuel on a local scale
- ❑ Determine the potential for expansion of the initial outpost
- ❑ Explore Mars – Understand Similarity to, and Differences from, Earth
    - Life – Past and Present
    - History of Atmosphere/Climate
    - Geologic Evolution and Present State

    Crew is assumed fit on Mars arrival

    Back contamination assumed resolved by precursor missions

Mars Exploration Program
- Mission Class Considerations -

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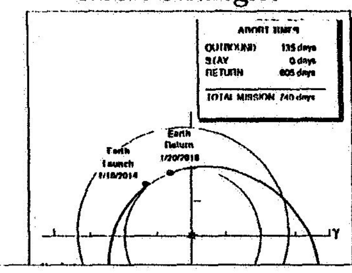
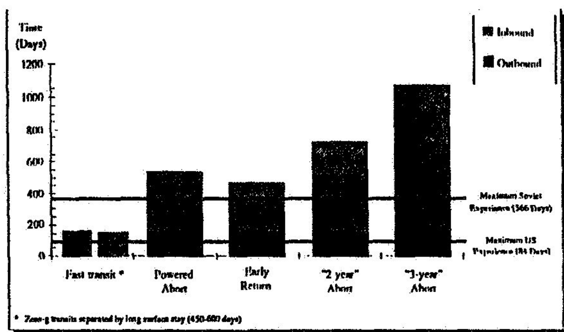
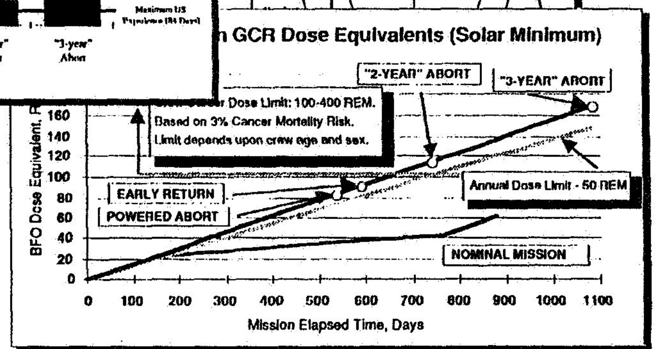
Abort Strategies





Zero-g Considerations



Galactic Cosmic Radiation Exposure

Mars Exploration Program
- Reference Mission Groundrules -

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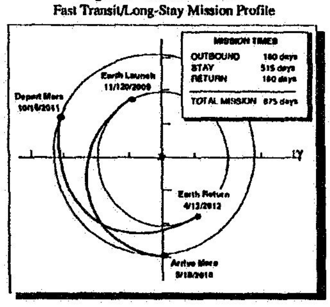
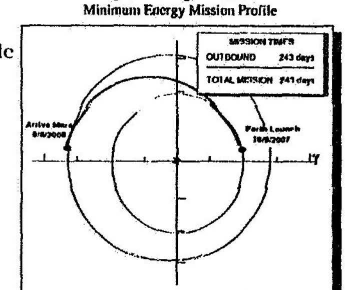
Figure 1 – Human Missions
Fast Transit/Long-Stay Mission Profile



- ❑ Split Mission Strategy:
  - Basic approach – humans on fast, moderately energetic transfers; cargo and all other assets delivered to Mars via minimum- energy trajectories
  - Eliminate LEO Assembly
- ❑ Human missions employ long duration stay (~550 days at Mars) mission profiles with fast (<180 days) Earth-Mars and Mars-Earth transit legs
- ❑ Abort strategy:
  - Aborts for human missions post-TMI are to the surface of Mars
  - Program assets directed toward the focus of the mission

- ❑ First human mission in 2010
  - Most challenging opportunity in the 15-year Earth-Mars cycle
  - Performance margin exists for other opportunities
  - Achievable development schedules
- ❑ First cargo mission in 2008
  - Cargo requirement of ~150 t to the surface of Mars
- ❑ Crew of 6
  - Based on past studies of skills mix and threshold psychological group dynamics
  - Reasonable starting point

Figure 2 – Cargo Missions
Minimum Energy Mission Profile



Mars Exploration Program
- Split Mission Strategy -

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❑ Objectives:

- Eliminate LEO Assembly for both cargo and piloted missions
- Use the FLO HLLV, or a FLO-evolved HLLV (shroud)
- Reduce number of HLLV launches
  - Send all surface and orbital assets to Mars on minimum energy trajectories
  - Crew-only use medium-energy, fast transit trajectories
- Provide mission flexibility to recover from contingencies
- Reduce engine testing requirements, if NTR is employed
- Provide launch window flexibility
  - 3/4 launches within the Earth-Mars window

❑ Results:

- First human mission to Mars reasonably achievable in 6 total launches of a FLO HLLV. Potentially achievable in 4.
- Significant mission content. 150 t of usable payload delivered to the surface of Mars
- No LEO assembly, rendezvous, or loiter needed
- Significant mission flexibility with this type of strategy

Mars Exploration Program
- Mission Overview -

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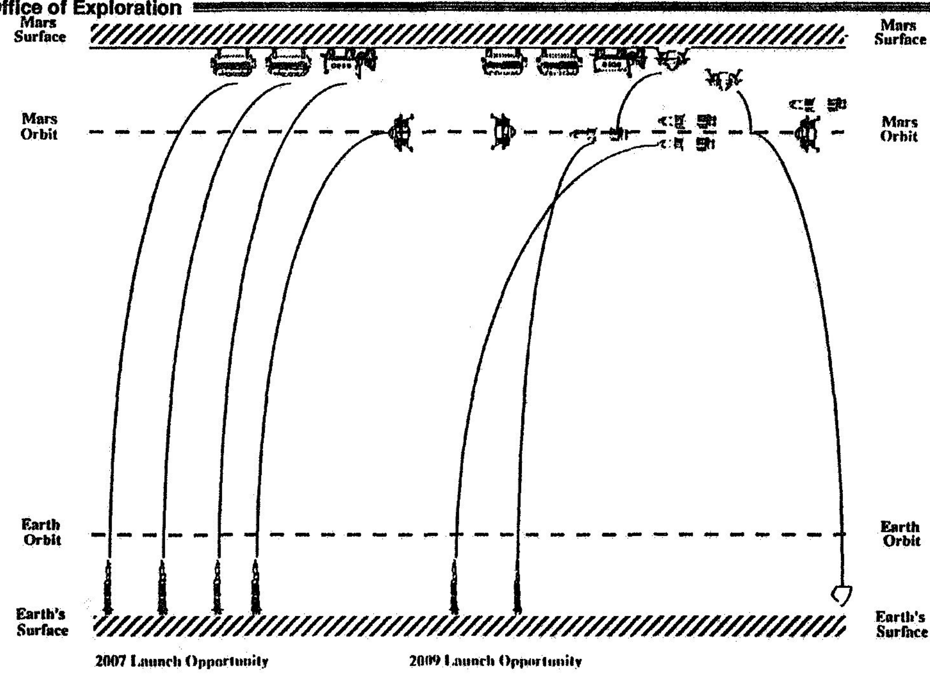

Mars Exploration Program
– Operations Concept –

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- Vehicle Prime
  - Realtime Systems Management
- Crew Prime
  - Realtime Exploration Functions
  - Crew Health Maintenance
  - Daily Planning and Resource Mgmt.
  - Preventive/Unscheduled Maintenance
- Crew Backup
  - Realtime Systems Management
- Ground Prime
  - Supplying Mission Objectives
  - Sustaining Engineering
  - Mission Critical Software Reconfigure
  - Crew Training
  - Procedures Development/Verification
  - Uncrewed Operations
- Ground Backup
  - Contingency Support
- Ground and Crew Share
  - Exploration

Mars Exploration Program
– Mission Transportation Elements –

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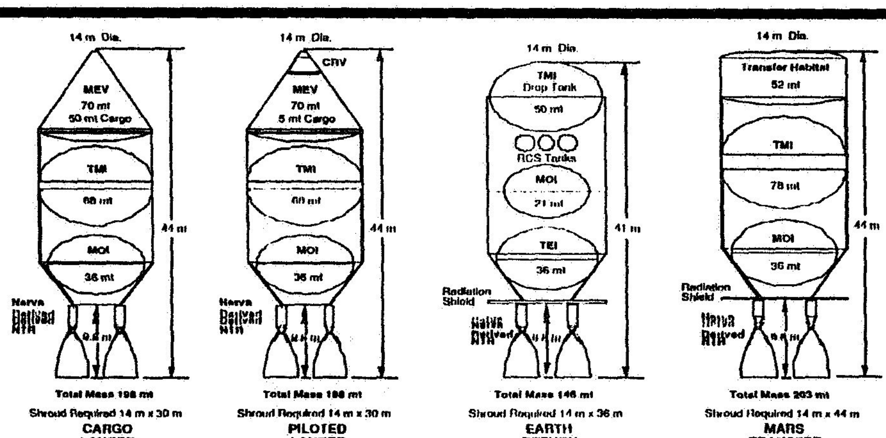
Mars Transfer Stage System Groundrules

- **NTR Propulsion (2 Engines-50 klbs. Thrust Each)**
- **Transit Habitat for 6 Crew / 360 Days**
- **Lunar HLLV Derivative**
- **No Radiation Disk Shield for Cargo Missions**
- **Mars Orbit - 250 km x 1 Sol Elliptical**
- **Separate Power Generation for Transfer Hab**
- **Automated Rendezvous for Mars Orbital Ops**
- **Storable RCS System for Vehicle Elements**
- **MEV for 6 Crew and 5 mt to Surface**
- **Zero-g Mars Transit**
- **Direct Entry Capability at Earth Return**



Mars Exploration Program
– Launch Vehicle Considerations –

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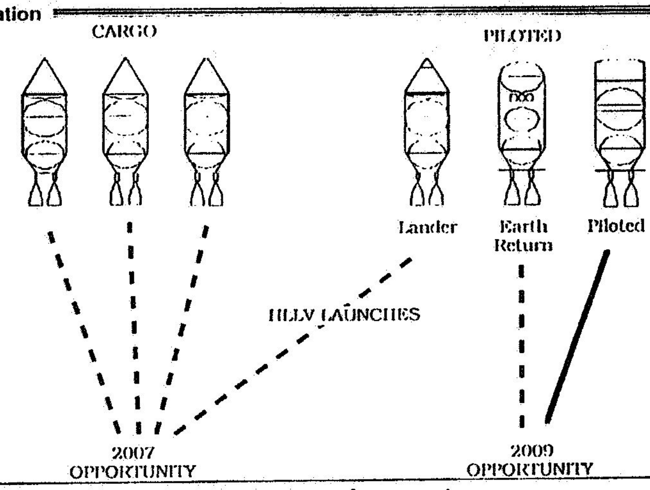

HLLV Requirements

- HLLV Derivatives – FLO or Early HLLV
- 200 mt Class IMLEO Launch Capability
- Shroud Size Options (Cylindrical Section): 14m x 30m or 10m x 50m
- Launch Window – ~90 days: 2-4 Launches per Mars Opportunity

Mars Exploration Program
– Mars Program Schedule (2007 Cargo Launch) –

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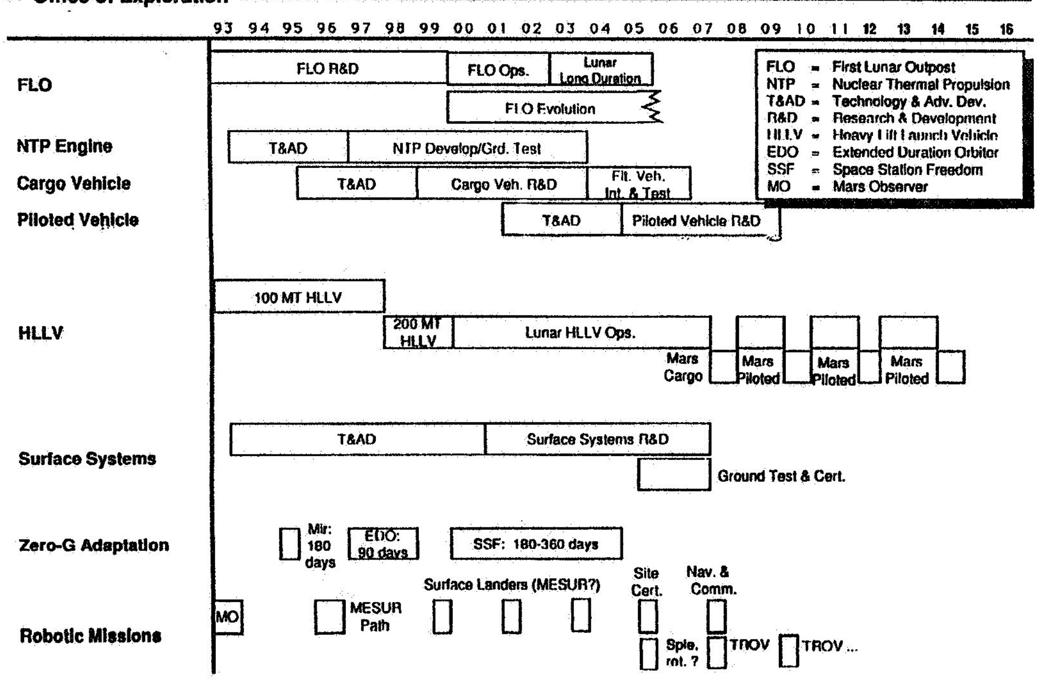


- Continue ExPO development of the Reference Mission
- Consider, compare, and contrast alternative reference mission concepts defined by non-ExPO teams
- Study system and subsystem implementation concepts to improve database

Solar System Exploration Division:
Requirements for Space Nuclear Propulsion

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 20, 1992

Douglas Stetson
NASA Headquarters

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION	Topics

- Solar System Exploration Goals and Missions
- Nuclear Electric Propulsion Rationale
- Nuclear Electric Propulsion Requirements
- Low-Power Missions
- Summary

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

Solar System Exploration Goals

- Solar System Origins
  - Understand the Process of Solar System Formation, in Particular Planetary Formation, and the Physical and Chemical Evolution of Protoplanetary Systems.
- Planetary Evolution and State
  - Obtain an In-Depth Understanding of the Planetary Bodies in Our Solar System and Their Evolution Over the Age of the Solar System.
- Evidence of Life
  - Search for Evidence of Life in Our Own and Other Planetary Systems, and Understand the Origin and Evolution of Life on Earth and Other Planets.
- Robotic and Human Exploration
  - Conduct Scientific Exploration of the Moon and Mars, and Utilize the Moon as a Base of Scientific Study in Participation with NASA's Mission from Planet Earth.

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

Next Mission Phases: Outer Planets

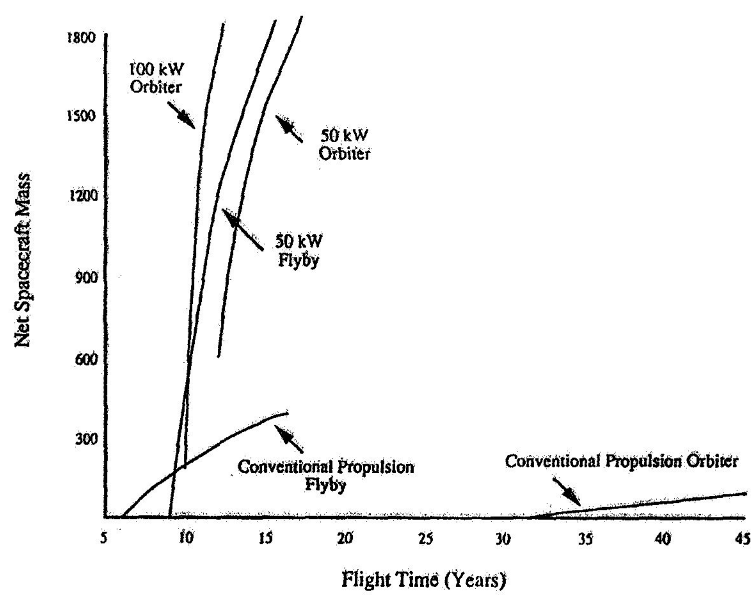
	Outer Planets					Other Planetary Systems
	Jupiter	Saturn	Uranus	Neptune	Pluto	Extrasolar
Reconnaissance	Pioneer 10 Flyby Pioneer 11 Flyby Voyager 1 Flyby Voyager 2 Flyby	Pioneer 11 Flyby Voyager 1 Flyby Voyager 2 Flyby	Voyager 2 Flyby	Voyager 2 Flyby	Flyby	Toward Other Planetary Systems
Exploration	Galileo Orbiter/ Jupiter Probe	Cassini Orbiter/ Titan Probe	Orbiter/ Probe	Orbiter/ Probe		
Intensive Study	Jupiter Grand Tour					

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION	Nuclear Electric Propulsion for Science Missions

- Nuclear Reactor Heat Source, Ion Propulsion System
  - Much More Efficient than Chemical Propulsion
- NEP Required for Next-Generation Outer Solar System Missions
  - Provides Payload Capability Unobtainable With Conventional Propulsion
  - Reduces Flight Time, Launch Vehicle Requirements
  - Also Enables High-Power Science Experiments
- SP100 Technology Baseline
  - Capable of 100KW for Outer Planet Missions
  - Lifetimes Up to 10 Years (Full Power)
  - Compatible With Active Power Conversion Technologies

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION	Benefits of NEP: Pluto Missions

PLUTO MISSION PERFORMANCE



SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

Jupiter Grand Tour

<u>Science Objectives</u>

- Thorough Characterization of Galilean Satellites
  - Geology, Morphology, Elemental Composition
  - Gravitational and Magnetic Properties
  - Interactions with Jupiter's Magnetosphere
- Follow-On to Galileo Study of Jupiter
  - Atmosphere, Inner Magnetosphere, Ring System

<u>NEP Mission Capabilities</u>

- Sequential Orbiting of All 4 Galilean Satellites
  - Comprehensive Imaging and Spectroscopy
  - Radar Sounding, Altimetry, Other Active Experiments
- Possible Addition of Jupiter Polar Orbiter or Satellite Landers
- Large Science Payload, ≈ 10 Year Mission Duration
  - Conventional Propulsion: 4 Separate Launches

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

Multiple Main Belt Asteroid Rendezvous

<u>Science Objectives</u>

- Comprehensive Study of Asteroid Physical Characteristics
  - Size, Shape, Density, Spin Properties
  - Surface Composition, Solar Wind Interactions
- Variations With Solar Distance
- Meaningful Sample Size, Variety of Spectral Types

<u>NEP Mission Capabilities</u>

- Rendezvous With 4-6 Main Belt Asteroids
  - Approximately 60 Days at Each Target
  - Possible Intervening Slow Flybys
  - Unlimited Orbit-Change Capability
- Large Science Payload
  - Imaging, Spectroscopy, Radiometry
  - Multiple Penetrators
- Total Mission Duration ≈ 10 Years
  - Conventional Propulsion: Max. 2 Targets, > 8 Years Duration

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

NEP System Requirements for Planetary Missions

Mission	Power Level (kWe)	Lifetime Full/Mission (Yrs)	Specific Mass (kg/kWe)	Technology Need Dates (year)
Far Outer Planet Orbiters/Probes	100	7-9/14-15	< 35	≈ 2000-01
Jupiter Grand Tour	100	8/11	< 35	≈ 2000
Multiple Mainbelt Asteroid Rendezvous	90	7/10	< 35	≈ 2000
Comet Nucleus Sample Return	90	4/8	< 35	≈ 2003

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

Low-Power NEP Missions

- Initial NEP System Will Address Reduced Requirements
  - Simplifies Development, Reduces Cost
  - Still Capable of Excellent Planetary Missions
- Mission/System Studies Ongoing
  - Joint NASA/DOE Report Issued
  - JPL/LeRC Study Focussing on Low-Power Missions
- Preliminary Mission Options Include:
  - Mars Orbiter, Phobos-Deimos Rendezvous (SEI Focus)
  - Main-Belt Asteroid Missions
  - Jupiter Satellite Mission
  - Solar Probe
- System Requirements (Preliminary):
  - Minimum 20 kWe NEP System
  - Minimum 3 Year Lifetime (Full Power)
  - Growth Potential to 100 kWe, 10 Years Lifetime

SSED REQUIREMENTS FOR SPACE NUCLEAR PROPULSION

Summary

- Nuclear Electric Propulsion Enables Next-Generation Outer Solar System Mission
- Requirements
  - 100 kWe, 10 -yr. Lifetime (Full-Power), < 35 kg/kWe
  - Initial System: > 20 kWe, > 3 Yr. Full-Power Lifetime
  - Full-Power System Launch ~2005

N93-26911

SPACE NUCLEAR THERMAL PROPULSION

SNTP

DOD REQUIREMENTS FOR SPACE NUCLEAR THERMAL PROPULSION

PRESENTATION TO

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

BY

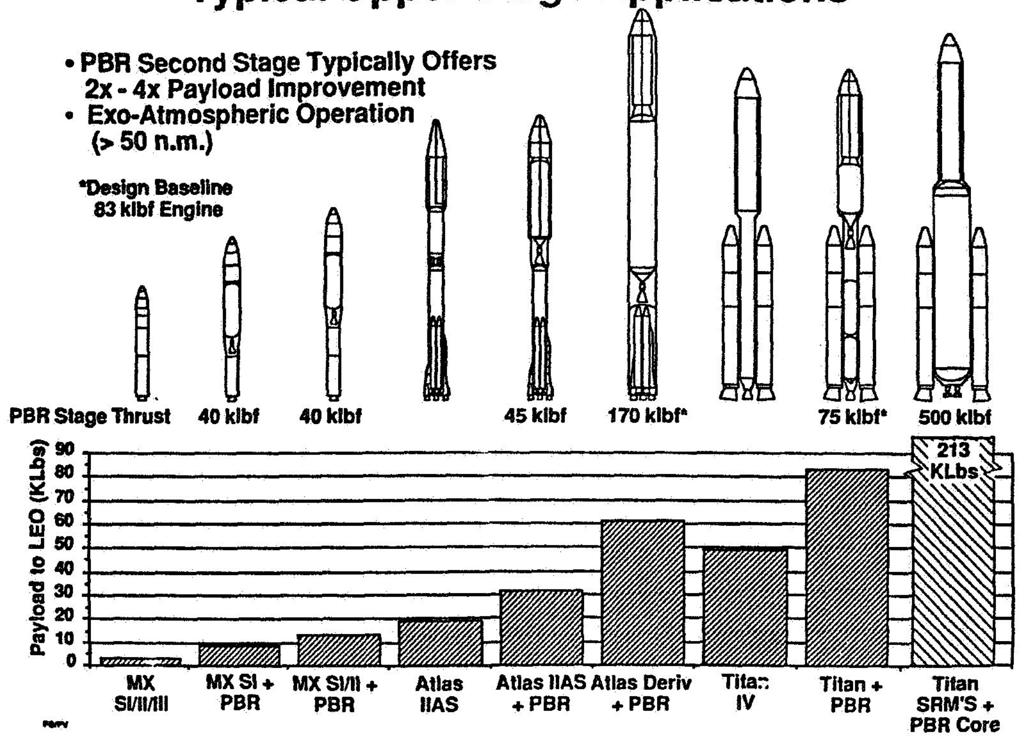
LT COL GARY A. BLEEKER
SNTP PROGRAM MANAGER
PHILLIPS LABORATORY

20 OCTOBER 1992

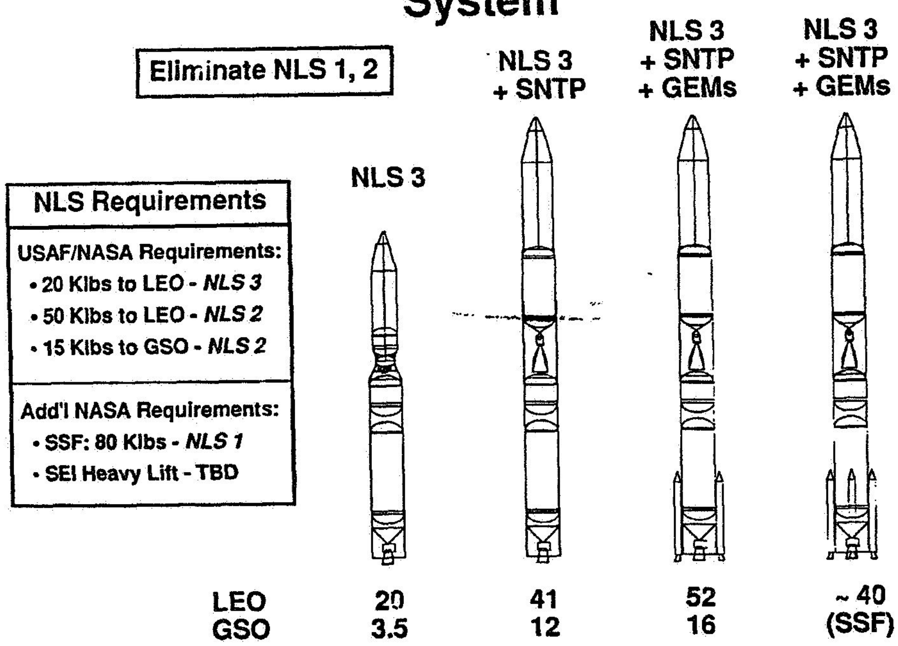
POTENTIAL DOD APPLICATIONS OF NUCLEAR THERMAL PROPULSION

- o UPPER STAGES ON EXISTING AND/OR NEW LAUNCH SYSTEMS
- o ORBIT TRANSFER VEHICLES (OTVs)
- o REUSABLE OTVs
- o ORBIT MANEUVERING VEHICLES

Typical Upper Stage Applications



Complement National Launch System



DOD APPLICATIONS NO LONGER UNDER CONSIDERATION

- o BALLISTIC MISSILE INTERCEPTOR SECOND STAGE
- o ICBM SECOND STAGE

DOD/AIR FORCE NTP REQUIREMENTS

- o DOD AND AIR FORCE DO NOT SPECIFICALLY CALL OUT NEED FOR NTP
  - -- CALL OUT MISSION REQUIREMENTS, NOT TECHNOLOGY
  - -- NTP COULD ENABLE MISSION ACCOMPLISHMENT (LAUNCH UPPER STAGE) AT LESS EXPENSE AND WITH GREATER RELIABILITY

SNTP PERFORMANCE GOALS

SNTP HAS THE FOLLOWING PERFORMANCE GOALS IN DEVELOPING AN ENGINE TECHNOLOGY WITH TWICE THE SPECIFIC IMPULSE OF H_2/O_2 ENGINES WITH COMPARABLE THRUST TO WEIGHT

THRUST:	**20,000 to 80,000 LB_F**
THRUST TO WEIGHT RATIO:	**UP TO 35 TO 1**
SPECIFIC IMPULSE, I_{SP}:	**1,000 SEC**
GAS CHAMBER TEMPERATURE:	**3,000K**
RUN TIME DURATION:	**1,000 SEC**
ENGINE CYCLES:	**3 TO 10**
ENGINE STARTUP TIME:	**UNDER 10 SEC**

Potential Cost Benefits

Assumed $1000/Lb Launch Cost to LEO (Past Year 2000)

Mission	Impact of SNTP	$/Mission *Non-Recurring	#/Year	20 Year Total	
National Launch System	Eliminate Large Core	$25 M + $2 B*	4	$4.0 B	
Atlas Upgrade	Titan IV Payload Capability	$130 M	4	$10.4 B	USAF $19.4 B
Orbital Maneuvering Vehicle	Retrieve/Repair High Value Satellites	$500 M	1	$5.0 B	

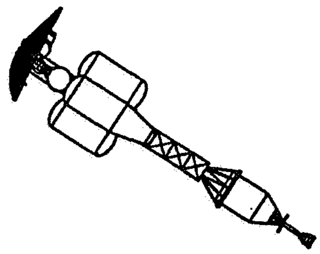
INTRODUCTION

EXECUTIVE SUMMARY

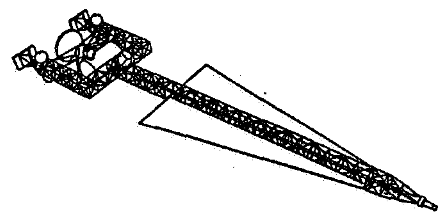
N93-26912

Focused Technology: Nuclear Propulsion

Nuclear Thermal Propulsion



Nuclear Electric Propulsion



Presentation to SSTAC/ARTS

Thomas J. Miller
10/21/92

NASA — LEWIS RESEARCH CENTER

OBJECTIVE

OBJECTIVE

DEVELOP AND DEMONSTRATE TECHNOLOGY FOR NUCLEAR PROPULSION SYSTEMS TO SATISFY USER CODE MISSION REQUIREMENTS

- BALANCE TECHNOLOGY AND PERFORMANCE WITH SOUND SAFETY AND ENVIRONMENTAL POLICIES

SCOPE

- NUCLEAR THERMAL
- NUCLEAR ELECTRIC

CUSTOMER

- LUNAR/MARS EXPLORATION (OEX)
- ROBOTIC SCIENCE (OSSA)

ELEMENTS

- CONCEPT DEVELOPMENT AND SYSTEMS ENGINEERING
- INNOVATIVE TECHNOLOGY
- ENABLING TECHNOLOGY (NEP & NTP)
- FACILITIES
- SAFETY, QA AND ENVIRONMENT

NUCLEAR PROPULSION OFFICE

NASA LEWIS RESEARCH CENTER

MISSIONS CONSIDERATIONS

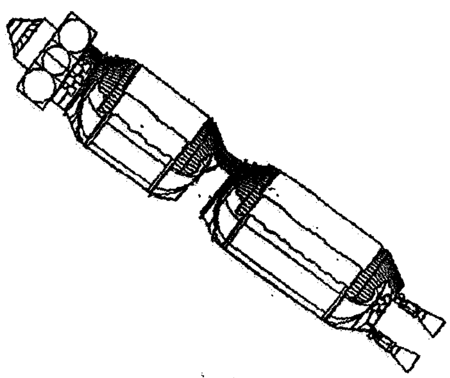
- SAFETY
- PERFORMANCE
- COST
- SCHEDULE FOR DEVELOPMENT
- OPERATIONAL FLEXIBILITY
  - APPLICATION TO RANGE OF MISSIONS
  - EVOLUTIONARY GROWTH POTENTIAL

NUCLEAR PROPULSION OFFICE

NASA LEWIS RESEARCH CENTER

NUCLEAR PROPULSION SUMMARY

NUCLEAR THERMAL PROPULSION

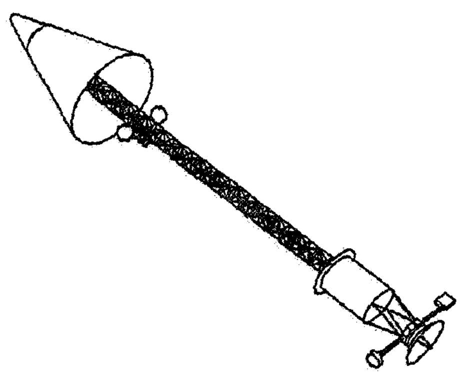


Specific Impulse*: 850 - 950 sec

Thrust to Weight: 6 - 10

$^*I_{sp} = T/\dot{m}$

NUCLEAR ELECTRIC PROPULSION



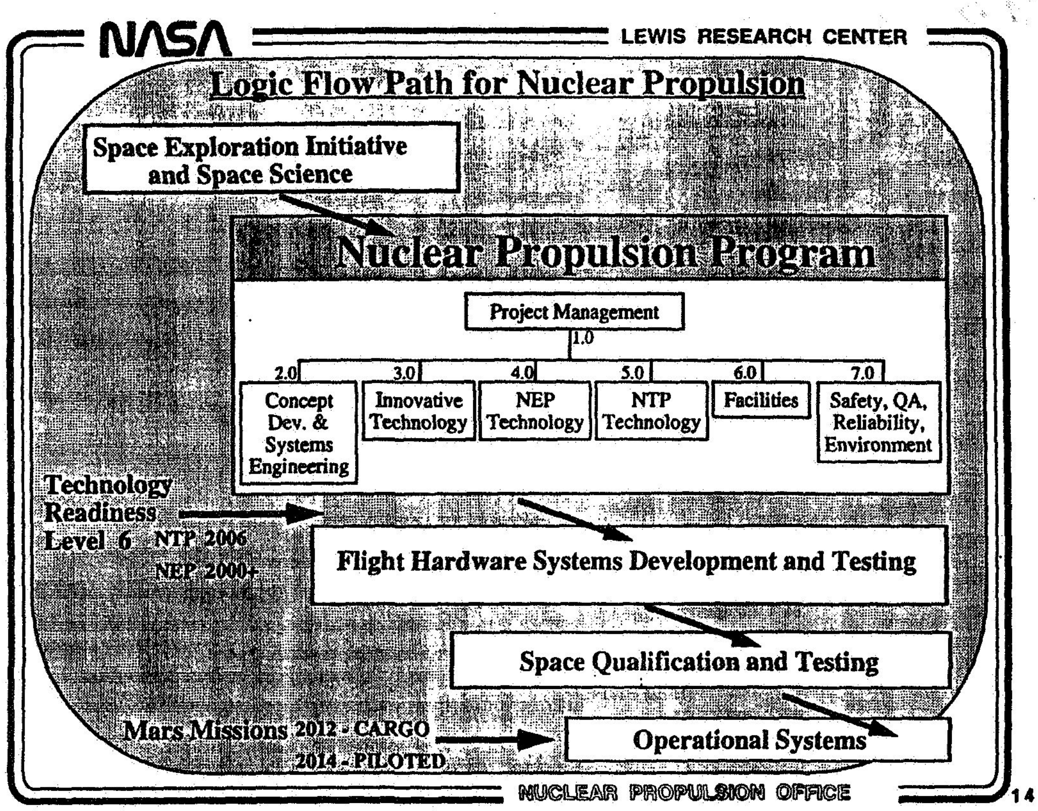
Specific Impulse*: 4000 - 8000 sec

Specific Mass:

Robotic Science	40 Kg/Kw_e
Piloted Mars	$\leq$ 10 Kg/Kw_e

CHEMICAL PROPULSION (H/O): 460 sec Specific Impulse

NUCLEAR PROPULSION OFFICE


NASA
LEWIS RESEARCH CENTER
Logic Flow Path for Nuclear Propulsion
Space Exploration Initiative and Space Science
Nuclear Propulsion Program
Project Management
1.0
2.0
Concept Dev. & Systems Engineering
3.0
Innovative Technology
4.0
NEP Technology
5.0
NTP Technology
6.0
Facilities
7.0
Safety, QA, Reliability, Environment
Technology Readiness Level 6 NTP 2006
NEP 2000+
Flight Hardware Systems Development and Testing
Space Qualification and Testing
Mars Missions 2012 - CARGO
2014 - PILOTED
Operational Systems
NUCLEAR PROPULSION OFFICE
14

N93-26913

OVERVIEW
OF
DOE SPACE NUCLEAR PROPULSION PROGRAMS

BY
ALAN R. NEWHOUSE
DEPUTY ASSISTANT SECRETARY
FOR SPACE AND DEFENSE POWER SYSTEMS
U.S. DEPARTMENT OF ENERGY

PRESENTED AT THE
NUCLEAR PROPULSION INTERCHANGE MEETING
NASA LEWIS RESEARCH CENTER, PLUM BROOK STATION
SANDUSKY, OHIO
OCTOBER 20, 1992

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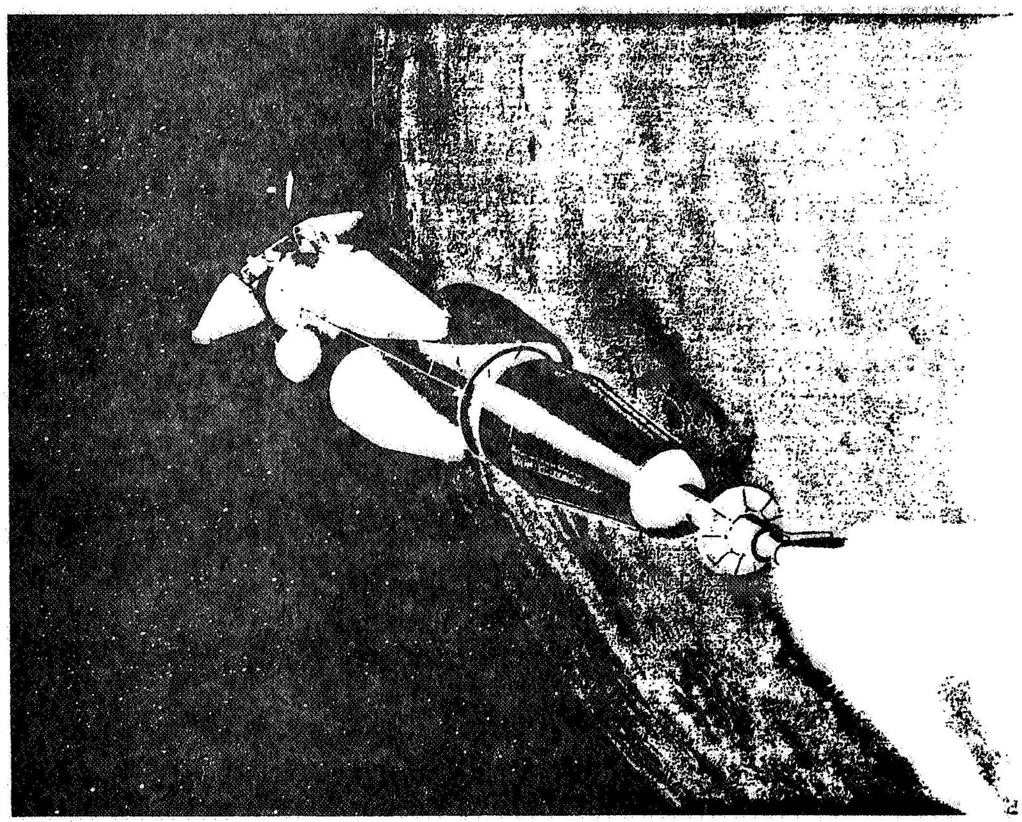


TYPES OF SPACE NUCLEAR PROPULSION

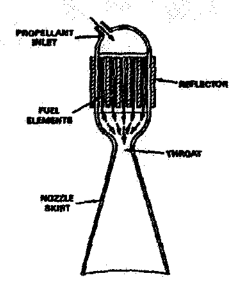
- NUCLEAR THERMAL ROCKETS (NTR)

  DEVELOP THRUST BY USING A NUCLEAR REACTOR TO HEAT A PROPELLANT GAS AND EXPEL IT THROUGH A NOZZLE

  - HIGH SPECIFIC IMPULSE* (>TWICE BETTER THAN BEST CHEMICAL SYSTEMS)
  - HIGH THRUST (FAST ACCELERATION)
  - SHORT LIFETIME (MINUTES TO HOURS)

  PRIMARY NASA OPTION FOR CARGO AND PILOTED MARS MISSION; ALSO, PROMISING CHOICE FOR MANY DOD APPLICATIONS

NUCLEAR THERMAL ROCKET



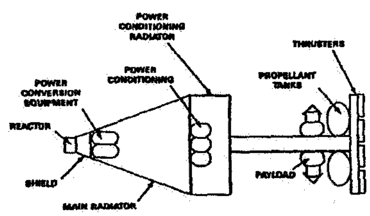
- NUCLEAR ELECTRIC PROPULSION (NEP)

  DEVELOP THRUST BY USING ELECTRICITY PRODUCED FROM HEAT IN A NUCLEAR REACTOR TO IONIZE A PROPELLANT AND ACCELERATE THE CHARGED PARTICLES THROUGH A THRUSTER

  - VERY HIGH SPECIFIC IMPULSE* (>10 TIMES BETTER THAN BEST CHEMICAL SYSTEMS)
  - LOW THRUST (CONTINUOUS ACCELERATION)
  - LONG LIFETIME (MONTHS TO YEARS)

  ENABLING OR SIGNIFICANTLY ENHANCING FOR SEVERAL NASA SOLAR SYSTEM ROBOTIC MISSIONS; NEAR TERM NEP REACTOR SYSTEMS WILL BE BASED ON SPACE REACTORS CURRENTLY UNDER DEVELOPMENT

NEP VEHICLE/SYSTEM SCHEMATIC



* THRUST PRODUCED PER RATE OF PROPELLANT CONSUMPTION

NUCLEAR THERMAL PROPULSION HISTORICAL SUMMARY (ROVER//NERVA)

- 18 YEAR DEVELOPMENT PROGRAM (1955-1973) ($1.4 BILLION EXPENDED IN THEN-YEAR DOLLARS)
- 20 REACTORS BUILT AND TESTED
  - REACTOR DESIGN AND DEVELOPMENT - LANL
  - DESIGN AND MANUFACTURE OF ROCKET ENGINE SYSTEMS - WESTINGHOUSE AND AEROJECT
  - TESTING OF ALL REACTORS - NEVADA TEST SITE
- PERFORMANCE DEMONSTRATED

- POWER LEVEL	4100 MWt
- PEAK FUEL TEMPERATURE	2750K
- SPECIFIC IMPULSE (Isp)	850 sec
- START/STOP CYCLES	28
- CONTINUOUS OPERATION	62 MINUTES

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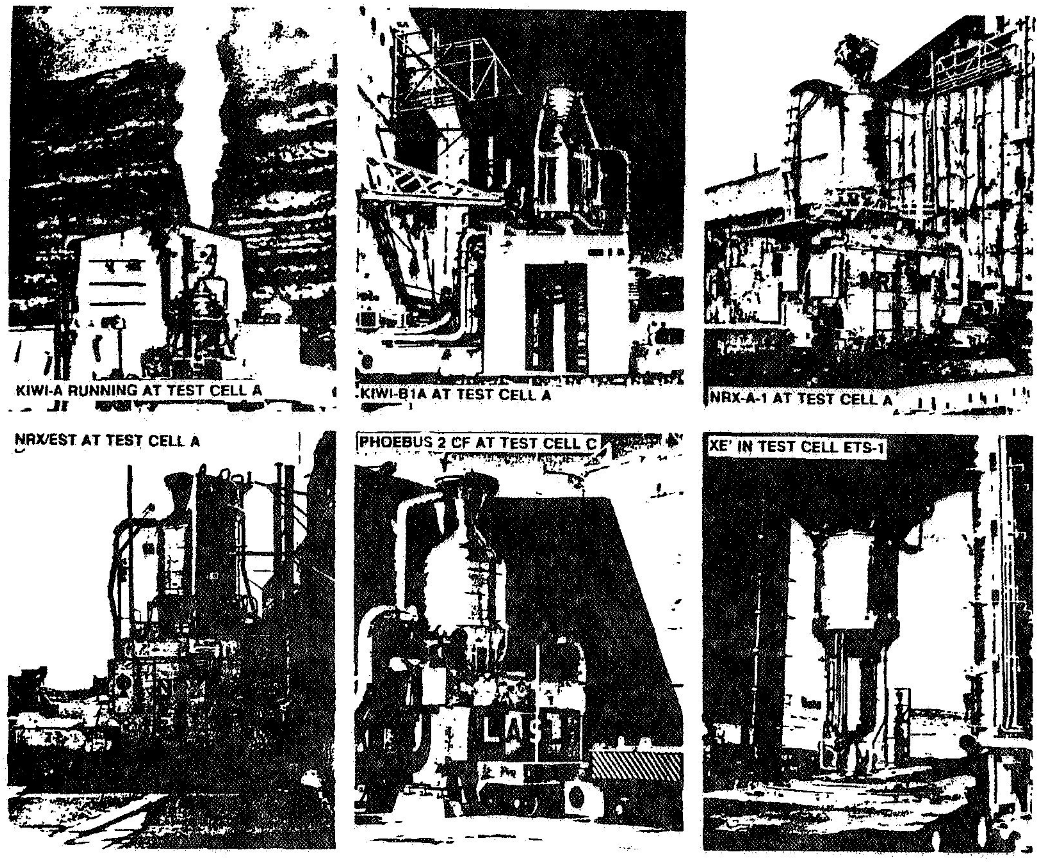

DOE'S CHARTER FOR THE DEVELOPMENT OF NUCLEAR POWER

- DOE'S CHARTER, ARISING FROM AUTHORITY IN THE ATOMIC ENERGY ACT OF 1954, AS AMENDED, IS TO SUPPORT FEDERAL AGENCIES (DOD AND NASA) IN MEETING THEIR SPECIAL POWER NEEDS FOR BOTH TERRESTRIAL AND SPACE APPLICATIONS

DOE ROLE IN SPACE NUCLEAR PROPULSION

- TRADITIONAL DOE ROLE OF DESIGNING, DEVELOPING, TESTING, AND PROVIDING NUCLEAR SYSTEMS, INCLUDING ENVIRONMENTAL, SAFETY, AND HEALTH ASPECTS
- NASA/DOE MEMORANDUM OF UNDERSTANDING (MOU) FOR ENERGY-RELATED CIVIL SPACE ACTIVITIES
  - SPECIFIC PROVISIONS FOR NUCLEAR PROPULSION
- PROJECT SPECIFIC MOU FOR NUCLEAR PROPULSION
  - DRAFT PREPARED FOR BOTH NASA AND USAF PROJECTS
- NATIONAL SPACE POLICY DIRECTIVE ON SPACE EXPLORATION INITIATIVE
  - NASA, DOD, AND DOE DIRECTED TO CONTINUE TECHNOLOGY DEVELOPMENT FOR SPACE NUCLEAR POWER AND PROPULSION

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DOE SAFETY ROLE IN SPACE NUCLEAR PROPULSION ACTIVITIES

- ENVIRONMENT, HEALTH, AND SAFETY
  - OVERALL POLICY, ALARA
  - OVERSIGHT
  - NEPA PROCESS
  - SAFETY ANALYSIS, REPORTS/APPROVALS
  - PUBLIC SAFETY
  - SAFEGUARDS
  - SITE MONITORING
- NUCLEAR SYSTEM DESIGN, MANUFACTURE, ASSEMBLY, CHECKOUT AND OPERATION
- GROUND TEST FACILITY DESIGN, ACQUISITION, CONSTRUCTION AND OPERATION

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SAFETY APPROACH

- SAFETY IS AN OVERRIDING CONSIDERATION:
  - FOR PROTECTION AGAINST ACCIDENTS
  - FOR PUBLIC ACCEPTANCE
  - FOR BOTH GROUND TESTING AND SPACE OPERATIONS
- ULTIMATE SAFETY OBJECTIVE:
  - MINIMIZE RISK TO PUBLIC AND CREW IN NORMAL AND ABNORMAL OPERATIONS
- NUCLEAR POWER SOURCE LAUNCH APPROVAL PROCESS:
  - BASED ON RIGOROUS SAFETY REQUIREMENTS, EVALUATION AND TESTING
  - CONSIDERS MISSION OBJECTIVES/BENEFITS VERSUS RISKS
  - BASED ON SUCCESSFUL HISTORY OF ISOTOPE AND REACTOR APPLICATIONS BY NASA AND DOD

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NUCLEAR SYSTEM DESIGN, MANUFACTURE, ASSEMBLY, CHECKOUT, AND OPERATION

- NUCLEAR SYSTEM DESIGN STANDARDS, REQUIREMENTS, AND CODES
  - DESIGN SAFETY FEATURES
  - SAFETY TEST REQUIREMENTS AND ANALYSIS
  - PREOPERATIONAL CHECKS AND TESTS
- MANAGEMENT OF FACTORY, SHIPPING, SITE, AND POST TEST OPERATIONS FOR NUCLEAR COMPONENTS AND SYSTEMS
  - FACTORY, SUB-ASSEMBLY TESTING, CRITICALS, ETC.
  - SHIPPING AND ASSEMBLY
  - FACILITY CONTROLS
  - EMERGENCY PLANNING
  - EMERGENCY ACTIONS
  - RECOVERY, CLEANUP, AND DISPOSAL ACTIONS
- NUCLEAR FLIGHT SYSTEM OVERSIGHT (WITH USER AGENCIES)
  - OVERALL POLICY DEFINITION
  - SAFETY REVIEW AND APPROVAL PROCESSES
  - FLIGHT OPERATIONS MONITORING
  - SUPPORT IN POSSIBLE EMERGENCIES
  - NORMAL AND ABNORMAL DISPOSAL OPERATIONS
  - POTENTIAL GROUND RECOVERY OPERATIONS

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GROUND TEST FACILITY DESIGN ACQUISITION, CONSTRUCTION, AND OPERATION

- SITE SELECTION AND MANAGEMENT OVERSIGHT
  - STANDARDS AND CRITERIA
  - SITE PREPARATION AND MAINTENANCE
  - SITE MONITORING
  - SHIPPING/HANDLING OF RADIOACTIVE/HAZARDOUS MATERIALS
  - DECOMMISSIONING AND DISPOSAL
- FACILITY DESIGN AND CONSTRUCTION OVERSIGHT
  - STANDARDS AND DESIGN CRITERIA
  - FUNCTIONAL REQUIREMENTS
  - SAFETY REQUIREMENTS, ANALYSES, AND APPROVALS
  - PREOPERATIONAL CHECKS AND TESTING OF EQUIPMENT AND SYSTEMS
- OVERSIGHT OF OPERATIONS
  - CONDUCT OF OPERATIONS
  - TRAINING REQUIREMENTS
  - TEST PROCEDURE APPROVAL
  - SPECIFIC TEST APPROVAL
  - POST IRRADIATION EXAMINATION
- QUALITY ASSURANCE PROGRAM OVERSIGHT
- SAFEGUARDS AND SECURITY OVERSIGHT

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GROUND TESTING ISSUES

- MAJOR FACILITIES REQUIRED
  - EITHER NEW FACILITIES OR EXTENSIVE MODIFICATIONS TO EXISTING FACILITIES
  - MUST MEET CURRENT ENVIRONMENTAL AND SAFETY REQUIREMENTS (EFFLUENT CONTROL)
  - FACILITY STUDY ESTIMATES $0.5 TO OVER $1B AND 7-10 YEARS EACH
  - DOE SITES WILL BE USED
- TYPES OF FACILITIES
  - FUEL BUNDLE QUALIFICATION
  - ENGINE SYSTEM
- ISSUES
  - SAFETY AND PUBLIC ACCEPTANCE
  - LARGE COST
  - SINGLE NATIONAL TEST COMPLEX VERSUS MULTIPLE COMPLEXES
- EXPERIENCE
  - ROVER/NERVA DESIGN
  - NUCLEAR FURNACE*
    - HAS SHOWN GROUND TESTING CAN BE ACCOMPLISHED THROUGH USE OF A SCRUBBER SYSTEM

*A SMALL (50 MWt) HIGH TEMPERATURE REACTOR USED FOR TESTING NUCLEAR THERMAL ROCKET FUEL ELEMENTS.

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POTENTIAL USE OF CIS FACILITIES

- TWO DOE DELEGATIONS RECENTLY RETURNED FROM A FACT FINDING TRIP TO RUSSIA AND KAZAKHSTAN
  - VISITED SEVERAL NUCLEAR PROPULSION FACILITIES WHICH COULD POSSIBLY BE USED
  - REVIEWED SPACE POWER AND PROPULSION CAPABILITIES
  - STILL COMPILING INFORMATION OBTAINED AND DRAFTING REPORTS
  - MUST CAREFULLY REVIEW AND VERIFY CAPABILITIES
- EXACT DOE ROLE IN USING OR MAKING USE OF FOREIGN NUCLEAR FACILITIES OR TECHNOLOGIES STILL NEEDS TO BE DEFINED
  - INTERNATIONAL AGREEMENTS MAY BE NEEDED
  - ROLE OF U.S. INDUSTRY NEEDS TO BE FURTHER EXPLORED
- TOPIC BEING WORKED

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NUCLEAR PROPULSION DEVELOPMENT NEEDS

- PAST PROGRAM PERFORMANCE NOT ADEQUATE FOR TODAY'S NEEDS
- PERFORMANCE IMPROVEMENTS REQUIRED
  - HIGHER SPECIFIC IMPULSE (900-1000 SEC.)
  - HIGHER THRUST/WEIGHT (25 - 35 TO 1)
  - DIFFERING REQUIREMENTS FOR CIVILIAN AND MILITARY APPLICATIONS (e.g., RUN TIME, RESTARTS)
- NEW DEVELOPMENT PROGRAM NEEDED
  - REESTABLISH OLD TECHNOLOGY AND CONSIDER NEW CONCEPTS

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KEY NEAR-TERM NUCLEAR PROPULSION ACTIVITIES FOR SPACE EXPLORATION

- DEVELOPING AND TESTING OF CANDIDATE NTP FUELS
- EARLY STUDY AND SELECTION OF EFFLUENT TREATMENT SYSTEMS
  - TEST AND QUALIFY PROTOTYPE COMPONENTS AND SUBSYSTEMS
- NUCLEAR FACILITY PRECONSTRUCTION ACTIVITIES
  - INITIATE ENVIRONMENTAL, SAFETY, AND PRELIMINARY DESIGN ACTIVITIES
  - PROCEED TOWARD A SINGLE NATIONAL NUCLEAR PROPULSION TEST COMPLEX
    - MEETS BOTH NASA AND DOD REQUIREMENTS
- NTP CONCEPTS ASSESSMENTS AND DEFINITION
  - NERVA DERIVATIVE
  - PARTICLE BED
  - CERMET
  - CIS TWISTED RIBBON

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RECENT DOE NUCLEAR PROPULSION ACTIVITIES FOR SEI

- LIMITED ASSESSMENTS OF NUCLEAR PROPULSION CONCEPTS AND ASSOCIATED TECHNOLOGIES
- NUCLEAR FUEL DEVELOPMENT
- FACILITIES EVALUATIONS AND ASSESSMENTS
  - DOE/NASA/USAF NUCLEAR FACILITIES REVIEW
    - INITIATED STUDIES FOR COMMON FACILITIES
  - PROVIDED FACILITIES INPUT FOR SNTP DRAFT EIS EFFORT
  - CIS FACILITIES VISITS
- PLANNING AND PROGRAMMATIC ACTIVITIES

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SPACE NUCLEAR THERMAL PROPULSION (SNTP) PROGRAM

- USAF TECHNOLOGY DEVELOPMENT PROGRAM TO DEMONSTRATE THE FEASIBILITY AND HIGH PERFORMANCE CAPABILITIES OF A NUCLEAR PROPULSION SYSTEM USING PARTICLE BED REACTOR TECHNOLOGY FOR POSSIBLE U.S. AIR FORCE (USAF) SPACE PROPULSION NEEDS
- USAF PHILLIPS LABORATORY IS PROGRAM MANAGER FOR THE SNTP PROGRAM
- DOE IS RESPONSIBLE FOR THE NUCLEAR DEVELOPMENT PORTION OF THE PROGRAM, INCLUDING NUCLEAR SAFETY OVERSIGHT AND NUCLEAR GROUND TESTING
- SANDIA NATIONAL LABORATORIES ALBUQUERQUE (SNLA) AND BROOKHAVEN NATIONAL LABORATORY (BNL) ARE PRINCIPAL DOE LABORATORIES PARTICIPATING ON THE PROGRAM

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SNTP NUCLEAR PROPULSION EIS ACTIVITIES

- DRAFT EIS
  - ISSUED FOR PUBLIC REVIEW
  - FINAL EIS EXPECTED IN NOVEMBER 1992
  - TWO SITES UNDER CONSIDERATION
    - -- NEVADA TEST SITE
    - -- IDAHO NATIONAL ENGINEERING LABORATORY TEST SITE
  - SITE SELECTION ANTICIPATED IN JANUARY 1993

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SUMMARY

- LONG HISTORY OF SUCCESSFUL USE OF NUCLEAR POWER IN SPACE (AND DOE SUPPORT OF THESE SYSTEMS)
- SPACE NUCLEAR THERMAL PROPULSION IS A LONG LEAD DEVELOPMENT ACTIVITY. CONSOLIDATION OF U.S. MILITARY AND CIVILIAN EFFORTS TO THE GREATEST DEGREE POSSIBLE WOULD BE BENEFICIAL.
- DOE WILL HAVE A LEAD ROLE IN DIRECTING THE NUCLEAR ASPECTS OF SPACE NUCLEAR THERMAL PROPULSION PROGRAMS; ACQUIRING AND OPERATING THE GROUND NUCLEAR TEST FACILITIES; AND ASSURING THE SAFETY OF ALL DESIGN, DEVELOPMENT, FABRICATION, TEST, AND OPERATIONS ACTIVITIES

5356 10/08/92

N93-26914

SPACE NUCLEAR THERMAL PROPULSION

SNTP

SPACE NUCLEAR THERMAL PROPULSION (SNTP) PROGRAM

PRESENTATION TO

NUCLEAR PROPULSION TECHNICAL INTERCHANGE MEETING

BY

LT COL GARY A. BLEEKER
PROGRAM MANAGER
PHILLIPS LABORATORY

20 OCTOBER 1992

SPACE NUCLEAR THERMAL PROPULSION PROGRAM

NUCLEAR ROCKET PROGRAM

- **TECHNOLOGY CHALLENGE**

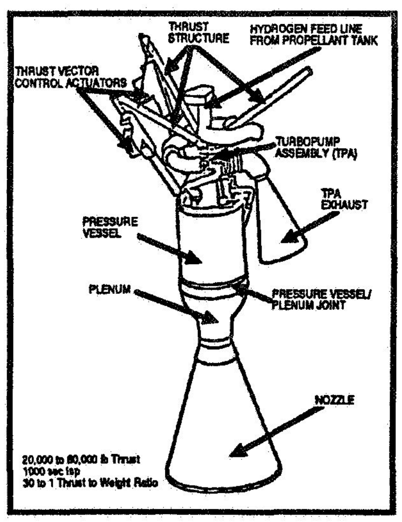
  -DEVELOP ADVANCED NUCLEAR ROCKET ENGINE WITH 2X THE ISP OF BEST LIQUID ENGINES AND THRUST TO WEIGHT COMPARABLE TO H2/O2

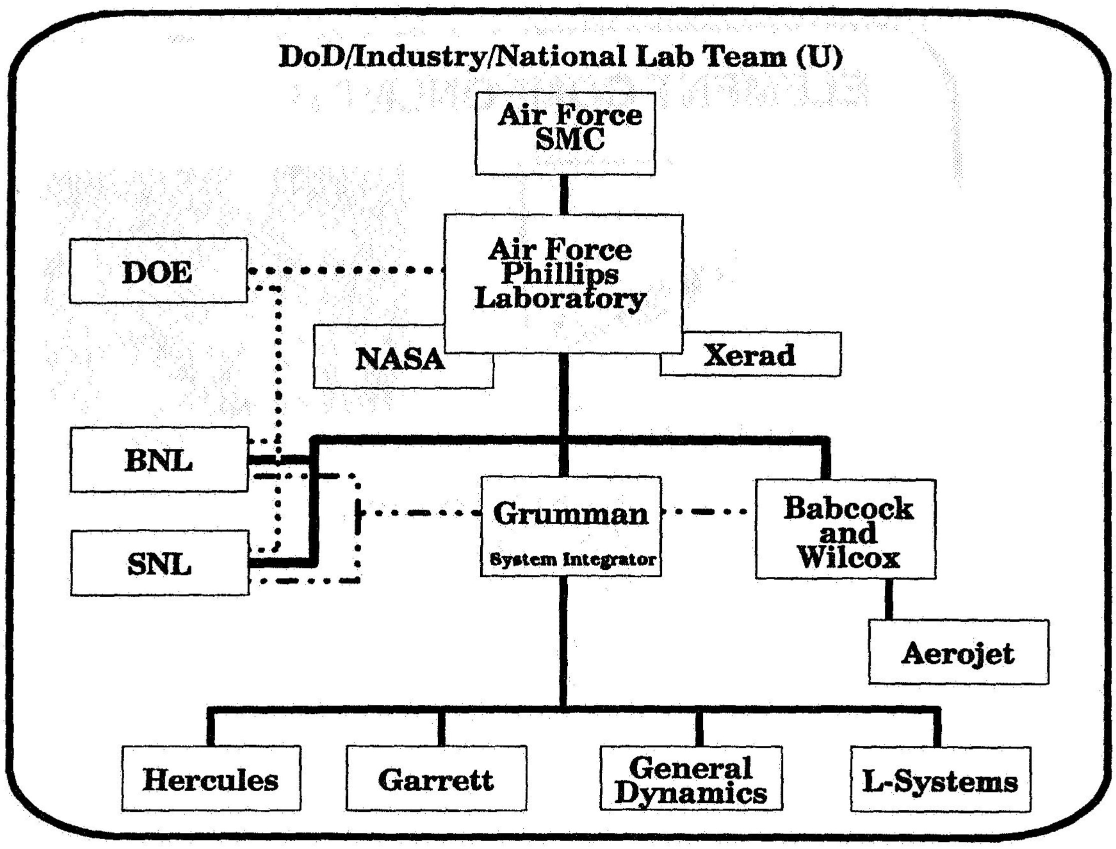
  -PROGRAM PRIORITIES ARE SAFETY, RELIABILITY, OPERABILITY, PERFORMANCE, AND AFFORDABILITY

- **PAYOFF**

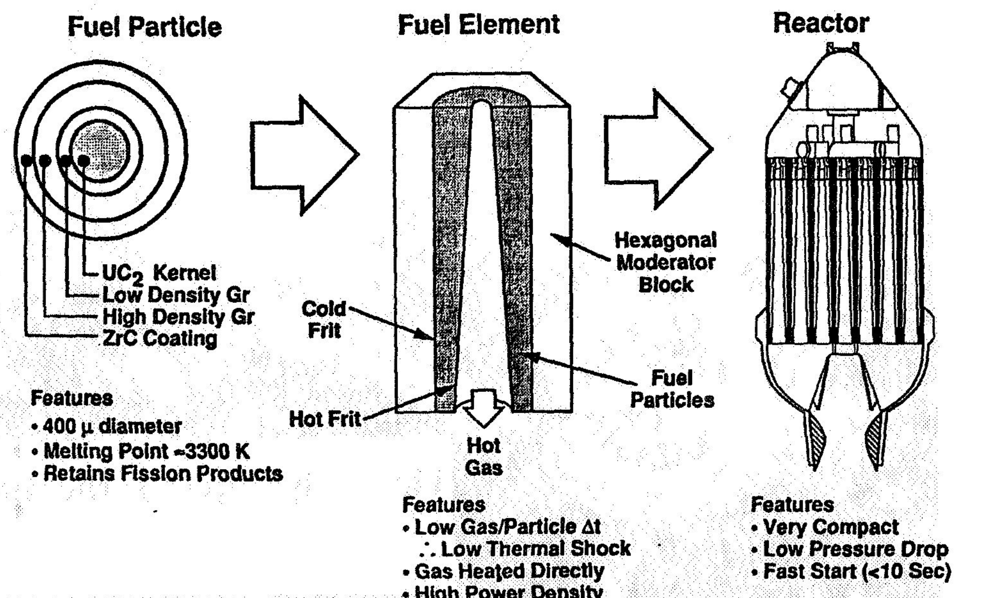
  -WIDE VARIETY OF POTENTIAL APPLICATION FOR UPPERSTAGES, OTV's AND PLANETARY MISSIONS

  -60-80% COST SAVINGS PER LAUNCH

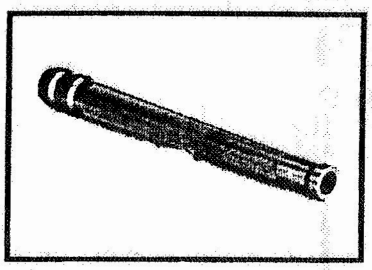




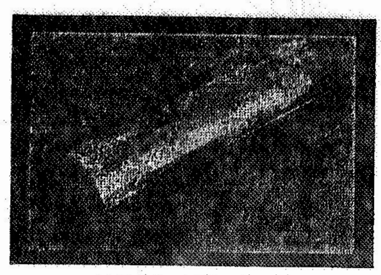
Enabling Technology - The Particle Bed Reactor



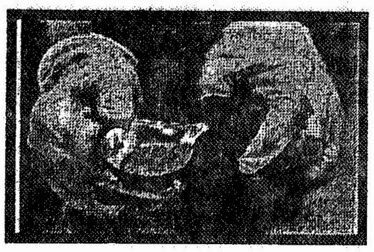
ELEMENT COMPONENT HARDWARE



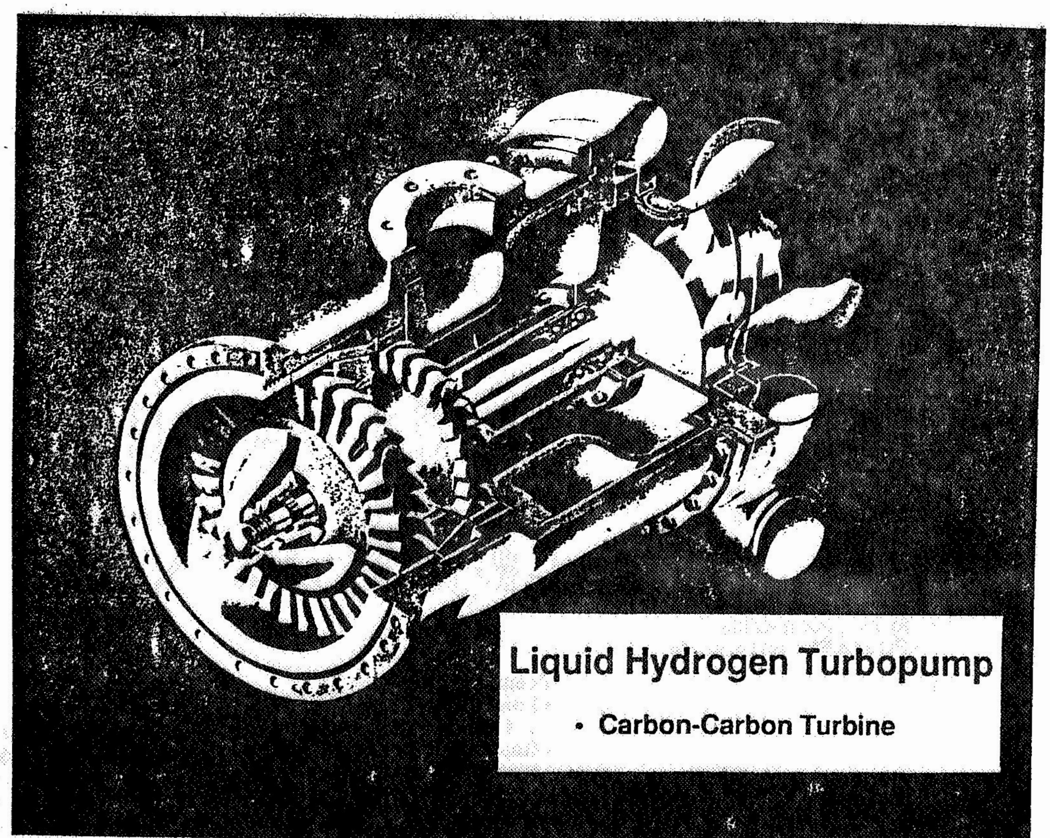
HOT FILTER



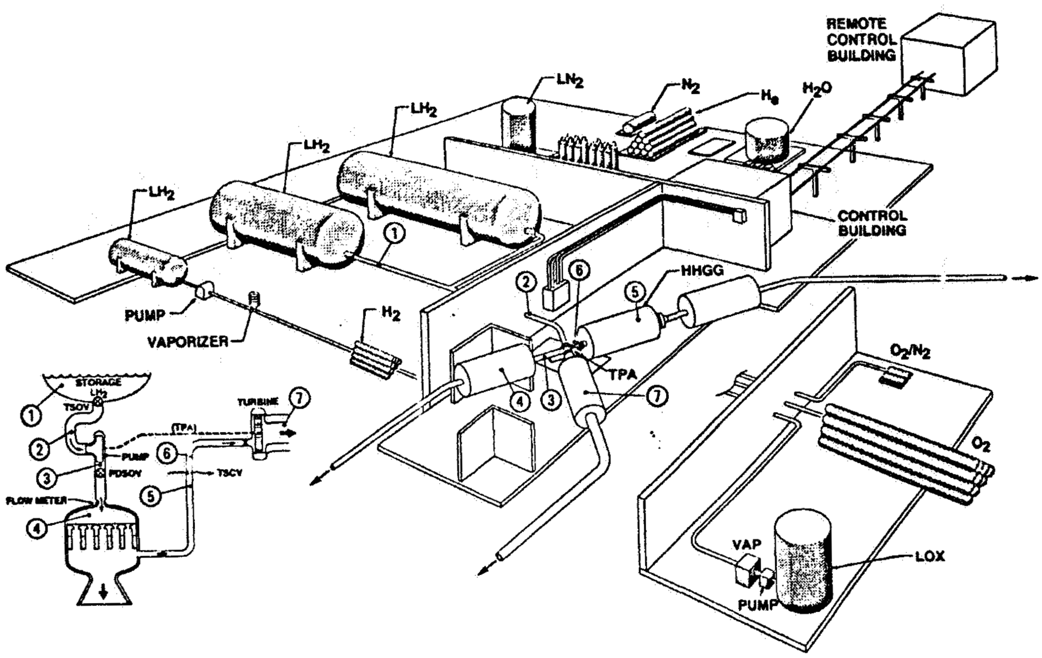
COLD FILTER



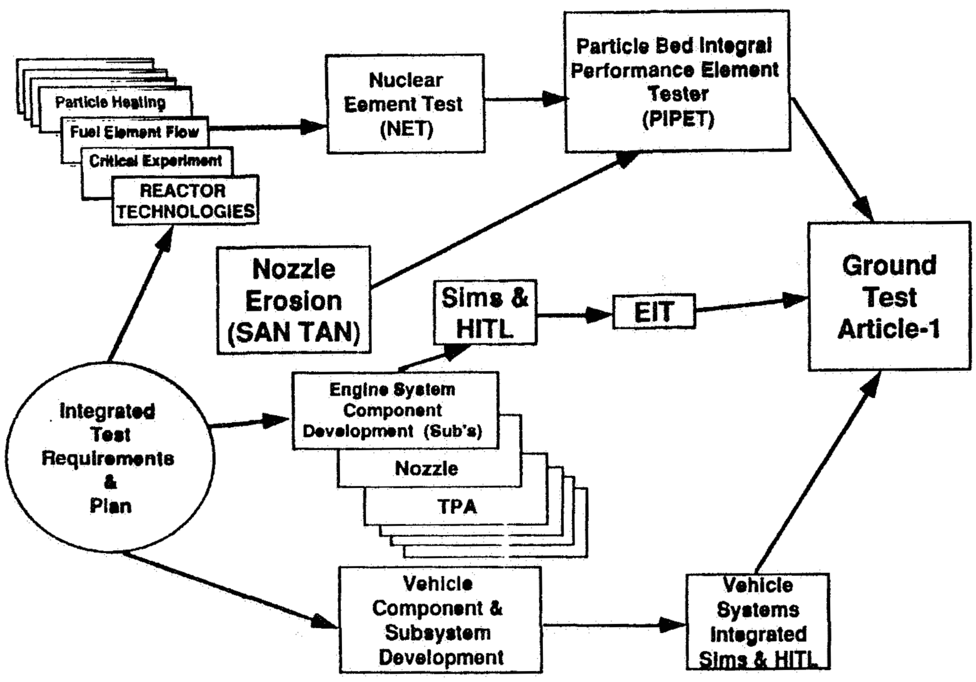
HIGH TEMPERATURE FUEL



SNTP Hydrogen Test Facility Layout

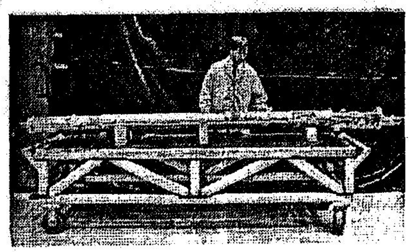


INTEGRATED TEST PLAN



NUCLEAR ELEMENT TESTING

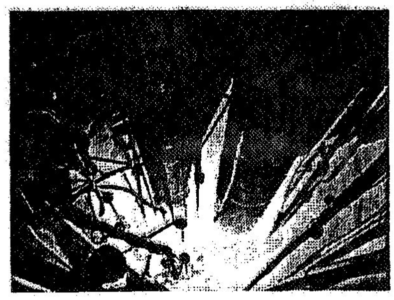
NET TEST CAPSULE



STATUS:

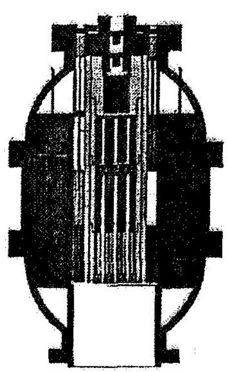
NET-0	COMPLETED (Non-Nuclear Checkout)
NET-1	Mar 93
NET-2	Sep 93
NET-3	FY94

ANNULAR CORE RESEARCH REACTOR



GROUND NUCLEAR TEST SITE

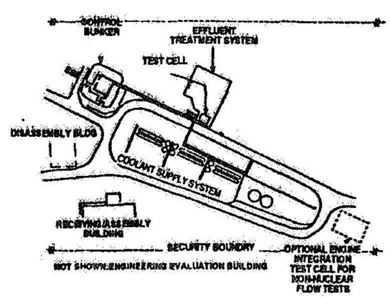
TEST REACTOR



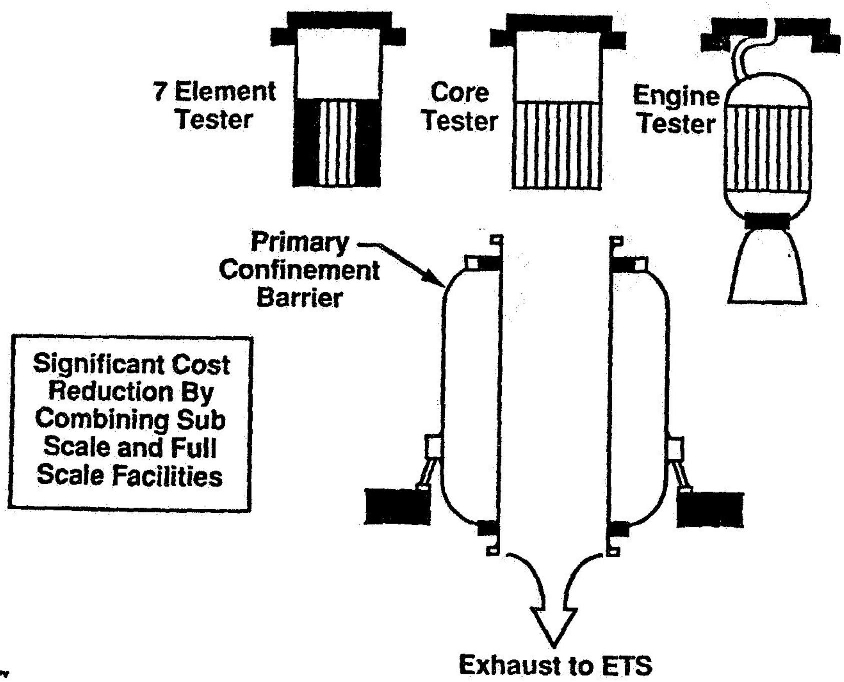
STATUS:

REACTOR PDR		Feb 92		
REACTOR FACILITY DESIGN	TITLE II	Feb92		
COOLANT SUPPLY SYSTEM	TITLE I	Oct 91	TITLE II	4Qtr FY93
EFFLUENT TREATMENT SYSTEM	TITLE I	2Qtr FY93	TITLE 1	FY94

GROUND NUCLEAR TEST FACILITY LAYOUT

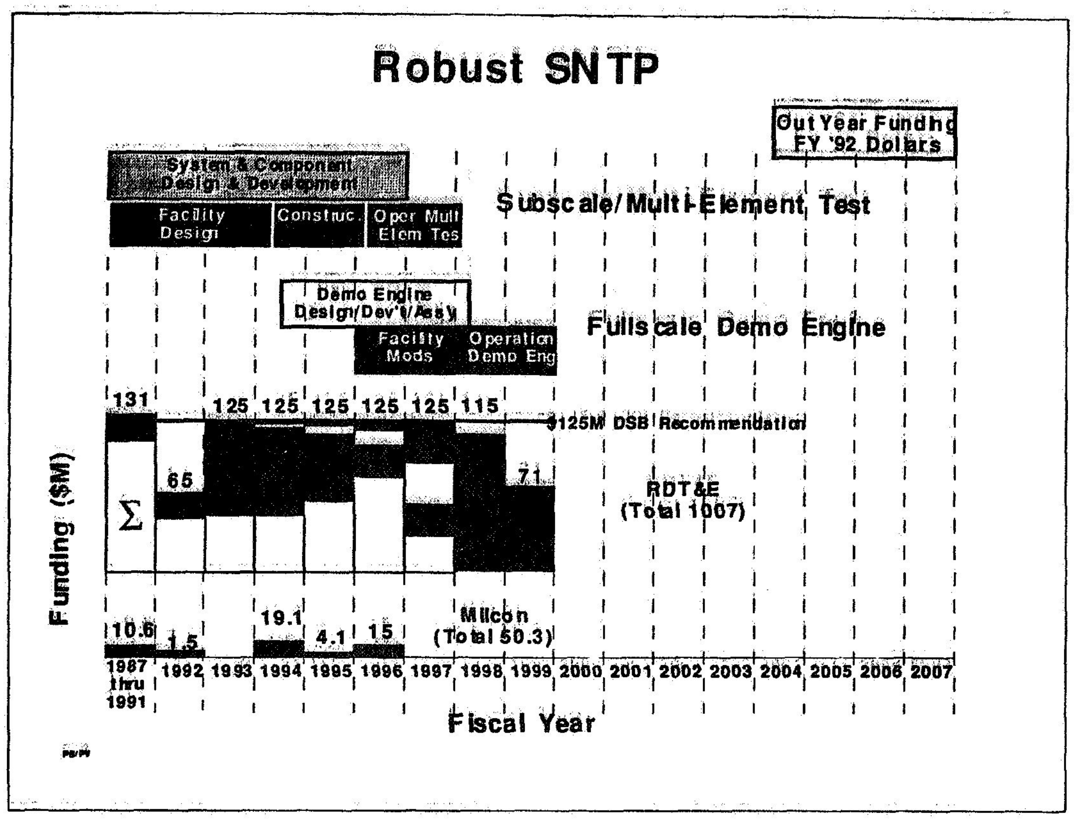


Ground Test Approach



SAFETY, ENVIRONMENTAL, HEALTH

- TOP PRIORITY FROM INCEPTION
  - PROGRAM SAFETY POLICY ESTABLISHED AND BEING FOLLOWED
  - PSAR COMPLETE AND UNDER REVIEW
  - MEETING ALL FEDERAL/STATE REGULATORY REQUIREMENTS
  - SUBSTANTIAL INTERNAL AND EXTERNAL REVIEW (DSB, DOE, NAS)
  - FOLLOWING ALARA (AS LOW AS REASONABLY ACHIEVABLE) APPROACH



SPACE NUCLEAR THERMAL PROPULSION

SNTP

PHILLIPS LABORATORY

SUMMARY

- NUCLEAR WILL BE THE PROPULSION SYSTEM OF THE 21st CENTURY
  - ESSENTIAL TO MAINTAIN U.S. COMPETITIVENESS AND SUPREMACY IN SPACE
- SNTP CONFORMS TO NATIONAL POLICY
  - HIGH PAYOFF R&D: MANY APPLICATIONS/MISSIONS
  - LEVERAGE DOD, DOE, AND NASA TECHNOLOGY BASE
- BASED ON CURRENT PROGRESS, PROGRAM HAS A HIGH PROBABILITY OF SUCCESS
- ALL APPLICABLE NUCLEAR SAFETY AND ENVIRONMENTAL OBJECTIVES WILL BE MET

NUCLEAR THERMAL PROPULSION

SYSTEM CONCEPTS

N93-26915

NASA LEWIS RESEARCH CENTER

Systems Overview

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 2,1992

Robert Corban
Nuclear Propulsion Office
NASA Lewis Research Center

NUCLEAR PROPULSION OFFICE

NASA — LEWIS RESEARCH CENTER

Systems Overview

Requirements and Public Acceptance

- OBJECTIVE
  - Provide NASA with Requirements management expertise
    - Requirement definition
    - Change management and control
    - Requirements document maintenance
  - Provide public acceptance planning
- Analytical Engineering Corporation
  - Awarded competitive contract (small business set-aside)
  - On-going 5 year contract
  - Provide key functional analysis
  - Initial requirements document developed and controlled

NUCLEAR PROPULSION OFFICE

Systems Overview
Requirements and Public Acceptance

The following charts provide a brief synopsis of the contracted efforts for FY92 in assessing Nuclear Thermal Propulsion requirements, concepts, and associated issues.

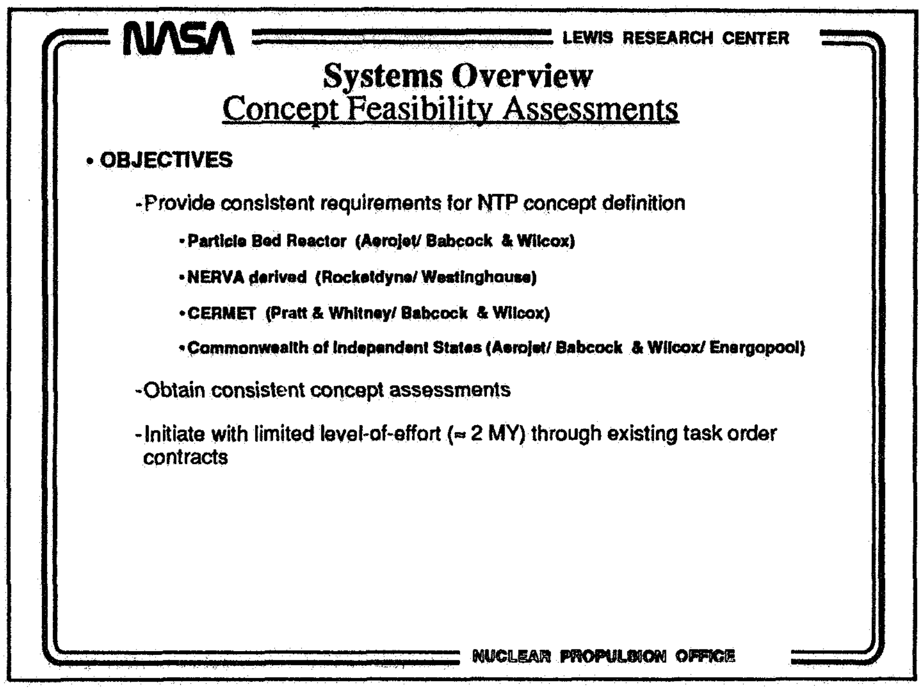
Requirements and Public Acceptance

Objective

This effort is to provide NASA LeRC with assistance in space nuclear propulsion system requirements management and public acceptance planning. Requirements management will include requirement definition, requirement change management and control, and requirement document maintenance. Specific objectives are to: 1) provide assistance in defining clear, concise, verifiable nuclear propulsion system requirements, 2) provide full traceability of requirements with reference, analysis, design, and historical data with the ability to assess the impact of requirement changes, 3) produce documentation of the nuclear propulsion system requirements and specifications that can easily accommodate changes, 4) provide assistance in public acceptance planning, and 5) include the resultant system requirements for a publicly acceptable SEI nuclear propulsion system.

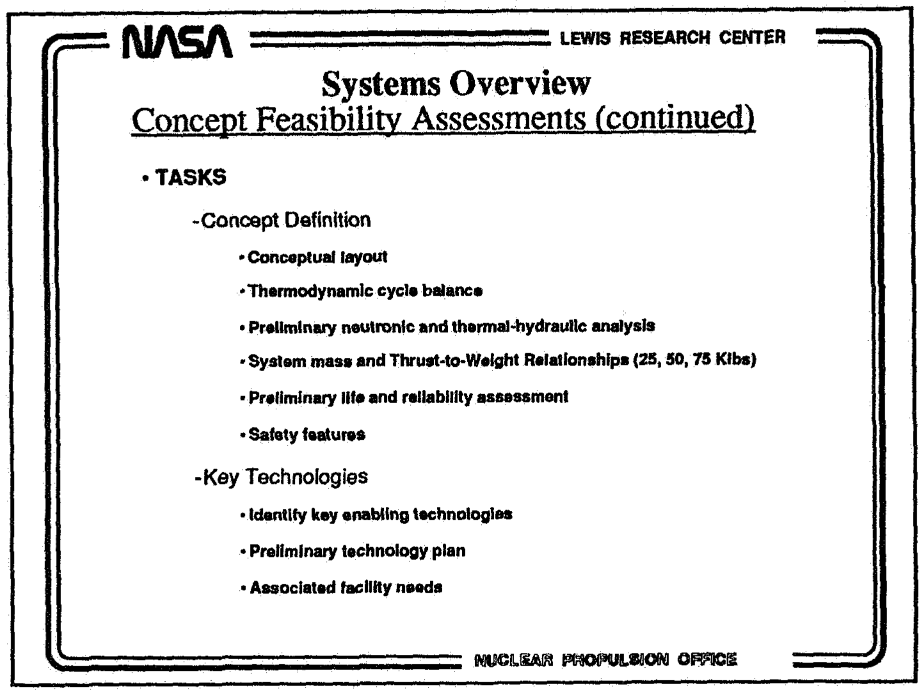
Analytical Engineering Corporation

Analytical Engineering Corporation (AEC) was awarded a five year contract in FY92 to meet the objectives defined above. AEC's approach will utilize detailed functional analysis to ensure that system functional requirements are accurately interpreted and flow down to system specifications. An initial requirements document has been developed and continuous improvements are on-going.



Systems Overview
Concept Feasibility Assessments

The objective of these studies was to determine the feasibility of a nuclear thermal propulsion system based on a particular fuel element form for the nuclear reactor. The studies evaluated "state-of-the-art" concept feasibility, thrust level range implications, test facility requirements, manned mission impacts, and key component technologies required. Shown in the chart are the study teams and their associated fuel element that was the basis for their concept analysis.



Systems Overview
Concept Feasibility Assessments (continued)

Concept Definition

The Contractors were requested to define a nuclear thermal propulsion concept based on their particular reactor concept in sufficient detail to permit reasonable judgements on feasibility, weight, performance, safety features, operations, and key technology requirements. An overall assessment of the NTP engine would include the reactor assembly, nozzle, propellant feed system, thrust vector control, instrumentation and control, and propellant pressurization. The concepts were defined to meet, as a minimum, the basic performance requirements defined below. The NTP engine concepts were assessed at one specific thrust level point with sensitivities determined for two others.

Baseline Design Requirements

PARAMETER	REQUIREMENT
Thrust	25K -75K
Thrust/Weight (w/ Internal Shield)*	≥4
Specific Impulse	≥850 seconds
Throttling	25% Thrust @ Rated Temperature
Reuse	Multiple (Mission Dependent ≥ 10 Restarts)
Single Burn Duration	60 minutes (Maximum)
Engine Life	>270 minutes at Rated Thrust (3X Required)
Reliability	Manned Systems
Propellant	Hydrogen

Key Technologies

The Contractors were to determine from the defined concept the key enabling technologies that need to be addressed before the system could be developed. Technologies that would have a significant impact on the overall system performance, safety, or reliability would also be identified. For each enhancing technologies, its system impact were to be identified along with the risks associated with its development.

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Systems Overview

- **Engine Clustering Study**
  - -Requested by EXPO
  - -Assess top level multiple NTP engine/vehicle clustering feasibility issues
  - -Determine impact on NTP requirements
  - -Contracted with General Dynamics
  - -Quick (2 months), Limited effort ($50K) study
- **Lunar NTR Vehicle Design & Operations Study**
  - -Identify and characterize "near-term" lunar transportation vehicles
  - -Assess design features, performance, and operational benefits
  - -Compare various lunar NTR options
  - -Contracted with SAIC and Martin Marietta
  - -One man-year effort over past six months

NUCLEAR PROPULSION OFFICE

Systems Overview

Engine Clustering Study

The objective of this study requested by NASA JSC's Exploration Project Office (EXPO) was to develop propulsion system designs that could be integrated with the provided reference vehicle. Four propulsion system options were developed using two and three engines with either boost pumps or run tanks for engine start up. The systems issues addressed consisted of TVC requirements, engine out possibilities, propulsion system failure modes and technology development requirements.

Lunar NTR Vehicle Design & Operations Study

The objective of this study was to identify and characterize the features of NTR propulsion stages for "near-term" lunar transfer vehicle missions. The study assessed NTR stage design features, performance, and operational benefits. Programmatic (schedule and cost) issues are also addressed. Comparison of various options for lunar transfer vehicles based on past studies on "all-propulsive" and "aerobraked" chemical were also addressed.

NASA LEWIS RESEARCH CENTER

Systems Overview

- **ENABLER I & ENABLER II**
  - Based on NERVA fuel element (scaled fuel for ENABLER II)
  - Parametric weight and size analysis
    - Thrust - Chamber Temperature - Chamber Pressure - Nozzle Area Ratio
  - Continued development of Nuclear Engine System Simulation (NESS) design program
  - Contracted with SAIC

NUCLEAR PROPULSION OFFICE

Systems Overview

Enabler I & II

The major objective of this task was to upgrade the Nuclear Engine System Simulation (NESS) analysis code to include the NERVA solid core engine (ENABLER I) and an advanced solid-core reactor module (ENABLER II) that utilizes scaled NERVA fuel elements. Additional objectives include the parametric characterization of the ENABLER I & II engine system concepts, and to examine on the "top-level" NTP engine design risk/reliability issues and their impact on the system.

N93-26916

REQUIREMENTS MANAGEMENT AND CONTROL

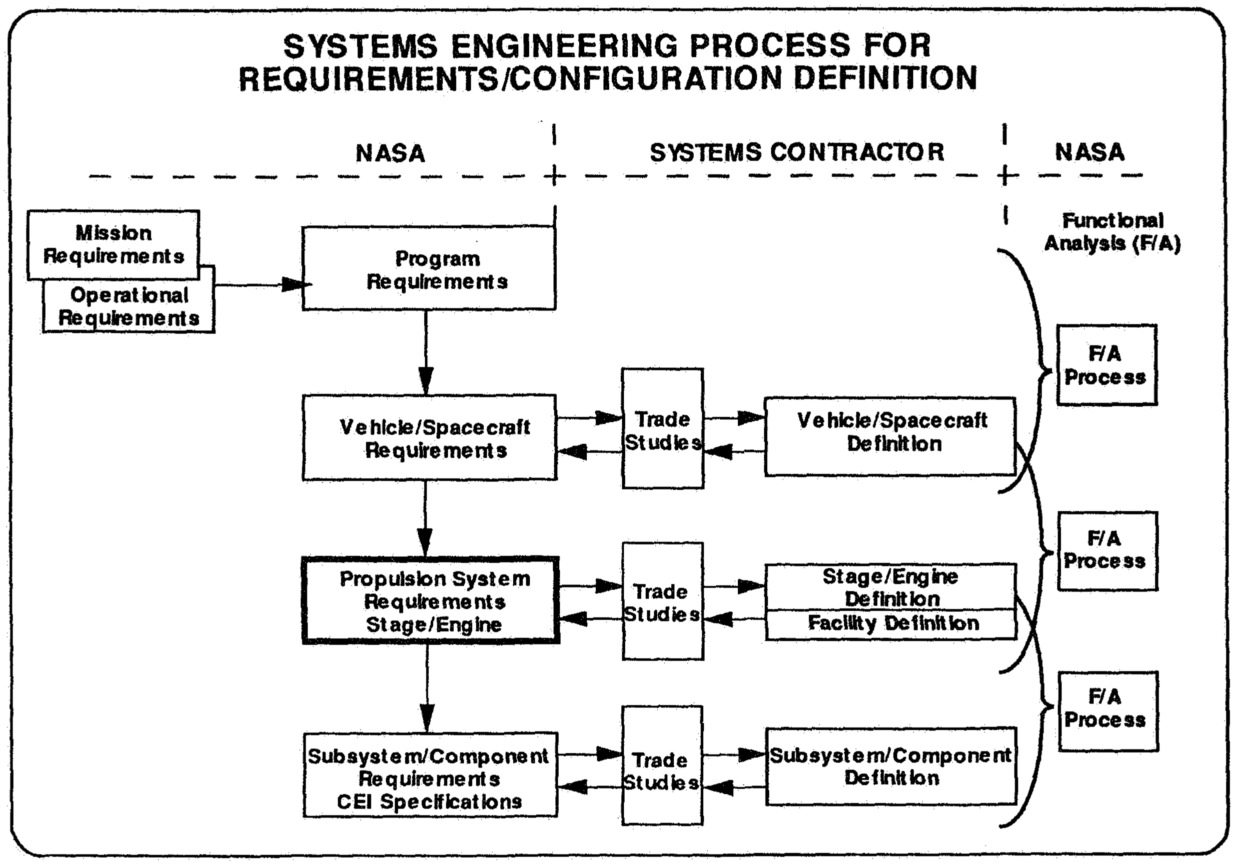
Presented by: Red Robbins

ANALYTICAL ENGINEERING

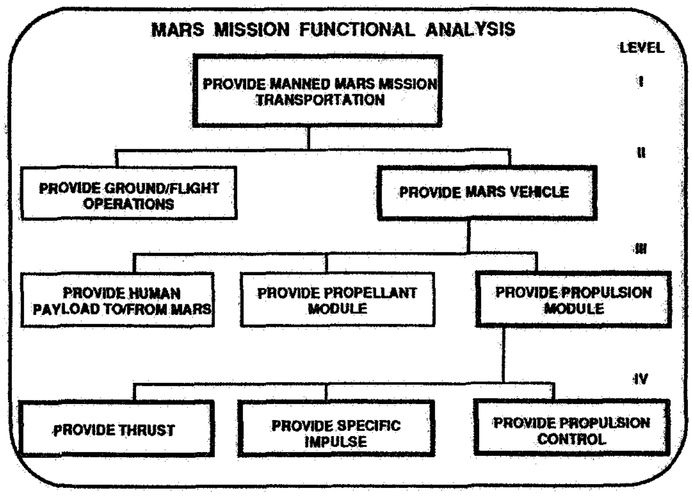
DISCUSSION

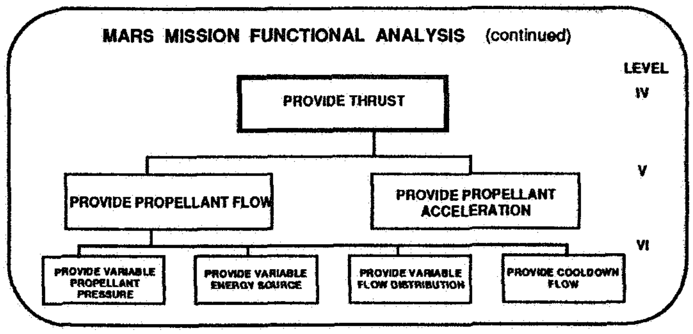
- Nuclear Thermal Propulsion Systems Engineering
  - Systems Requirements Status
  - Functions That Need To Be Performed
  - Attributes Associated With The Functions

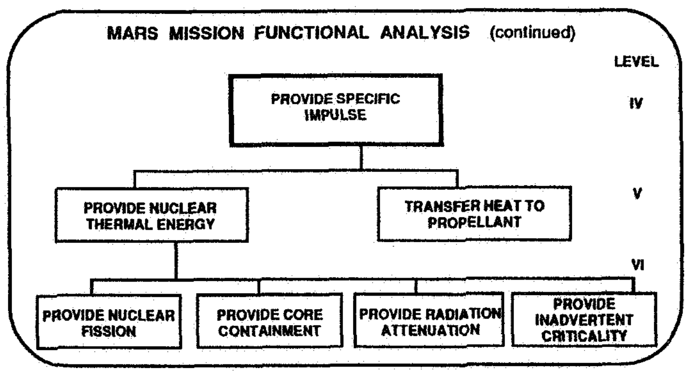
ANALYTICAL ENGINEERING

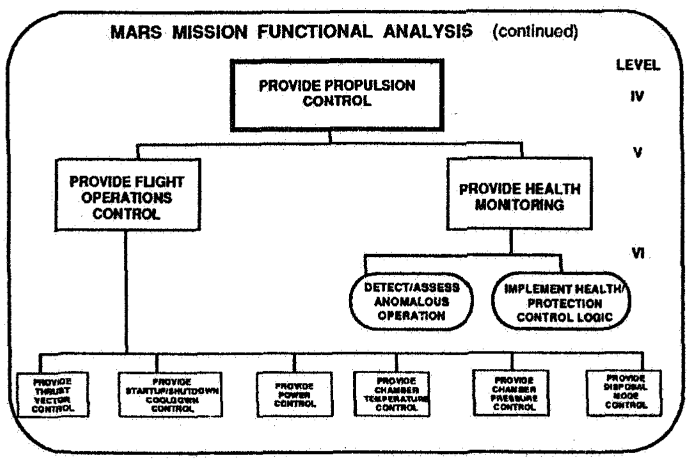


The systems engineering process for requirements and configuration definition, shown in the figure above, includes roles and responsibilities of NASA and the Systems Contractor. The overall program requirements are derived from the mission to be performed. The program requirements, in turn, are utilized as the basis for the definition of all lower level documents that contain increasing detail concerning design requirements associated with performance, safety, operations and environment. The lower level documents are dynamic in nature. Many studies are conducted involving trades between lower level requirements and engineering system definition. The propulsion system requirements are highlighted because they are the focus of this discussion. Stage/Engine requirements have been generated, baselined by NASA and are under formal change control and propulsion system definition is underway. The Functional Analysis activity shown is simply the process of systematically identifying the generic functions to be performed at all levels and leads ultimately to the definition of the System Architecture. Since Program and Vehicle Requirements are not currently available, the propulsion system requirements were generated on the basis of representative manned lunar and Mars missions.

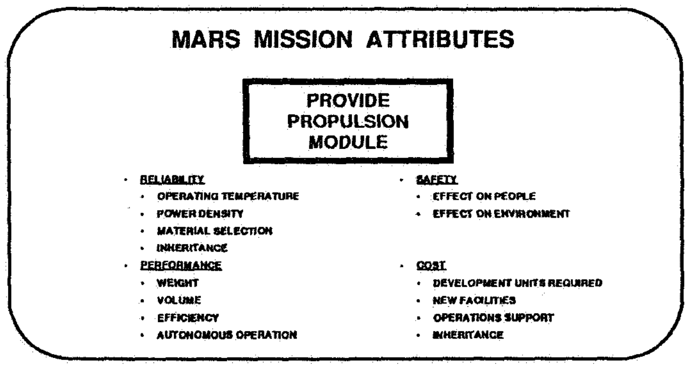


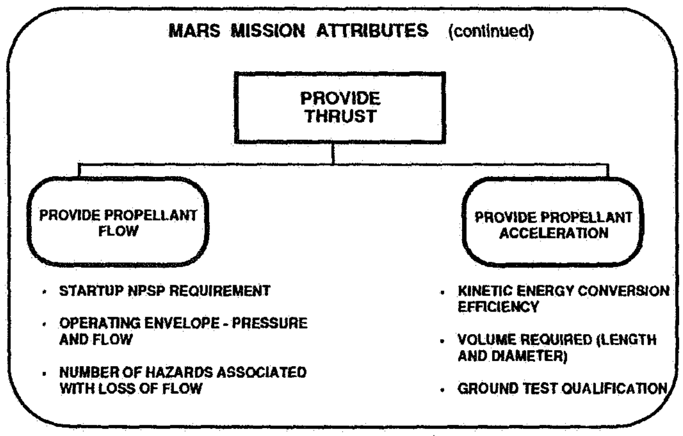


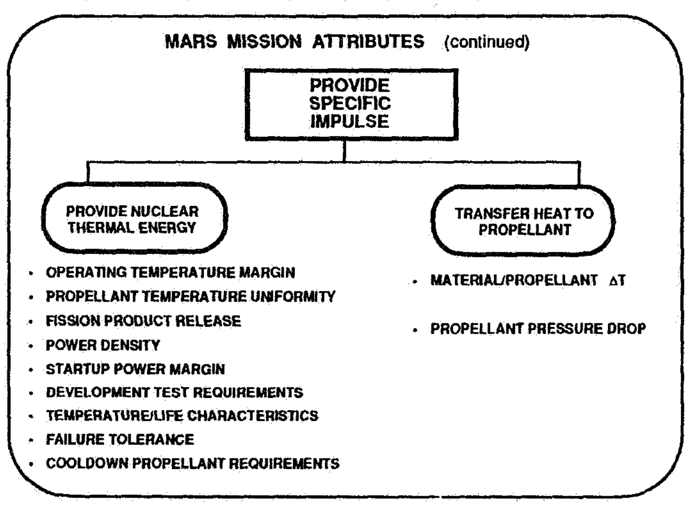


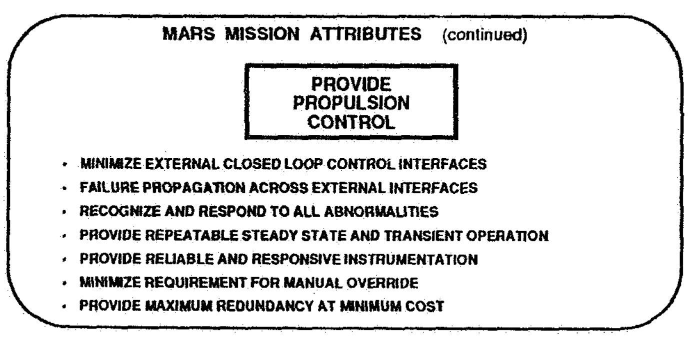


The Mars Mission functional analysis shown defines the mission functions to be performed in successive levels of detail. The highlighted functions at each level are those that are directly related to the propulsion system. Those functions that are not highlighted are the principal interfaces with the propulsion system. The three functions of the propulsion module are to provide thrust, provide specific impulse and control of the engine operations. The three charts which follow are simply functional breakdowns of the main engine functions to successively lower levels of detail. On the basis of this analysis, the system architecture or system definition can be completed. It should be noted that these functions are generic and are required for any nuclear rocket. No design solutions have been assumed. It is the role of the systems definition contractors to perform the system trades which result in the definition of the propulsion system.









The completion of the functional analysis previously described permits the definition propulsion system attributes that can be utilized to assess the relative merits of competing systems and to establish criteria for technology thrusts to enhance performance, safety and reliability. The attributes associated with the functions are outlined in the four charts that follow. These attributes, in addition to the system requirements, can be utilized for the evaluation of any nuclear rocket system and can provide the system contractors guidance about system characteristics that are considered to be important.

SUMMARY REMARKS

- A Consistent Set Of Propulsion System Requirements Has Been Developed By NASA
  - Traceable To Mission Needs
  - Under Formal Change Control
- Propulsion System Functions Traceable To System Requirements Have Also Been Generated
- Preliminary Propulsion System Attributes Traceable To Functions Have Been Derived To:
  - Assess The Relative Merits Of Competing Systems/Elements
  - Establish Criteria For The Definition Of Technology Thrusts
- We Request Feedback From The Program Participants - Particularly The System Definition Contractors
- Future Work Will Be Directed To The Development Of An Integrated Propulsion Systems Model Coupled With Propulsion Systems Requirements Traceable From The Lowest Level To Mission Needs

ANALYTICAL ENGINEERING

N93-26917

NUCLEAR SYSTEMS IN SPACE? DOES/WILL THE PUBLIC ACCEPT THEM?

Harold B. Finger

Public Acceptance is always raised as an obstacle to the use of nuclear energy for any purpose, in any way. It is always cited as an issue that must be resolved before nuclear energy can be used for:

Nuclear energy plants to generate more electricity,
Nuclear medical diagnosis and treatment,
Food irradiation to destroy harmful bacteria.

So it is not surprising that the assumption is generally made that there is public opposition to using nuclear energy in space that could preclude its use even for missions that it makes realistically feasible. Yes, there is a broad assumption that the public generally opposes nuclear energy.

Let me start right off by telling you that assumption is **WRONG.** (Figure 1) Here are some of the attitude data that indicate the public's attitudes on nuclear energy. They are positive, not negative. Most of the public believes nuclear energy will play an important role in our energy supply, that it should play an important role, and that the need for nuclear energy to supply our electricity will increase. Only 15% would favor closing our nuclear electric plant.

In spite of those data, you are not alone in thinking the public opposes nuclear energy. When (Figure 2) opinion leaders are asked how important a role they think nuclear energy should play in meeting our future energy needs, 72% answered Very or Somewhat important. But, then, when they were asked how important they thought the public feels about the reliance on nuclear energy, only 25% thought the public felt nuclear energy should play an important role, while 63% felt the public did not believe it should be important. As Figures 1 and 2 show, 73% of the public, the same number as the opinion leaders, believe nuclear energy should play an important role. A similar perception gap exits between Congressional staff views supporting the importance of nuclear energy and what they think the public believes.

So, (Figure 3) we all do have a job to get opinion leaders and our policy makers and many other influentials in our society to understand that the public accepts and even supports the use of nuclear energy. Doing that will certainly help get favorable policy action related to nuclear energy. But it won't be easy to get that point across. It won't be easy, at least partly because the small number of committed anti-nukes are vocal and because -- as about two thirds of those who call news about nuclear energy describe those news reports as negative -- the press does generally emphasize the negative. It appears that good news is not considered newsworthy.

As the USCEA has determined, based on broad attitude research (Figure 4), there should be no expectation that the public will accept or support the use of nuclear energy unless it meets special needs and offers special and significant benefits. That is why the USCEA's public information program emphasis (Figure 5) is on gaining recognition for the growing need for electricity in a growing economy and on nuclear energy's benefits in cutting imported oil dependence, reducing pollutant emissions and preserving scarce resources.

In transferring that lesson to our space use of nuclear energy (Figure 6), it means getting recognition and support for the space program broadly and for the missions that benefit substantially from or realistically require nuclear energy for their accomplishment.

This is what a group of aerospace and other companies are now trying to organize -- a program to do just that. If any of you here, whose organizations have not yet been involved in this effort want to become part of it, please let me or Red Robbins know of your interest. We'll welcome your participation.

Developing an effective public communication program (Figure 7) requires a solid base of attitude research. We must understand the views of the public and of our policy makers. We must determine those benefits of the space program and of the missions that are realistically enabled by nuclear energy that would be effective in gaining support for the space program and those missions. In fact, we know almost nothing about the public's attitudes and knowledge on using nuclear energy in space. I doubt that the public knows that we have already used nuclear -- radioisotope- power units in space to get data from the Moon in Apollo, to get pictures of Saturn and Jupiter, and other uses whose results were broadly and proudly discussed. We need to get such information known as part of our developing program.

We do have a fairly good feel for what the public thinks about the space program; thanks largely to the excellent work supported mainly by Rockwell International and from several others. So let me review some of those research results with you.

Here (Figure 8) are the generally highly positive views of the space program. Over 80% support the space program overall; believe it is important to the United States; approves of it; and, at least back in 1988, believed that a U.S. lead in the program was important. Figure 9 shows further data. There is less, though still strong, sense of a personal benefit than a national benefit, but it is certainly encouraging that relatively few- only 25 to 30 percent- considered space exploration a luxury at those times. I'll address that further later.

It is also important and encouraging to see the overwhelmingly positive responses when various benefits are suggested as reasons for supporting the space program (figure 10). However, all of these attributes are <u>suggested</u> in the interviews; there are no open-ended questions that would ask the interviewee what he or she knows and believes is most important about the space program. Of course, that will require further attitude research. In the meantime, the data of Figure 10 are very positive.

Here (Figure 11) are the responses when various goals are suggested for the space program. You'll notice that the support for all the proposed missions dropped from 1990 to 1992. We don't really know the reason for that drop, but it may also indicate that we have not adequately explained the economic, job, nor technology benefits of the space program. Even some Congressmen, who should know better, say we should not spend our budget IN space, that we need the work here on the ground. That's actually an argument we faced and addressed back in the 1960's. The response is obvious, I believe.

Although Figure 11 shows the significant downturn in support of manned lunar and Mars missions, let me turn to broader public views concerning the manned Mars mission, which we would all agree is certainly one of the primary missions for nuclear thermal propulsion. That mission is realistically enabled by nuclear propulsion.

For our Russian friends who are here, Figure 12 shows the obvious feelings of Americans that think we should do the Mars mission together with the republics of the former Soviet Union. Americans felt that way back in 1988 when we were strong

competitors. I expect the numbers would be much higher in favor of that joint effort today.

In essence, the various data here indicate that Mars and planetary investigation rate high among the alternatives suggested for future missions. Support for the President's SEI missions also shows high figures. However, it is significant that only a little over a third of those interviewed were aware of his proposals. That is only another manifestation of the fact that his initiatives were not broadly discussed and that they were not seized within the space community nor developed and pushed as dynamic goals that could provide significant benefits for the country. There was very little discussion of those goals and proposals outside the space and science community.

The question of the importance of the U.S. being first to get to Mars drew a response that, not surprisingly, change significantly after the demise of the Soviet Union and its replacement by the Commonwealth of Independent States. In 1989, there was a small margin feeling it was important that we be first, but after the Soviet coup attempt, there was a significant reversal with only 35 percent feeling it was important that we be first. The competition with the Soviet Union was no longer considered significant as a justification for an urgent effort to be first in that difficult Mars goal. As I indicated earlier, the idea of a joint effort may be viewed as an even greater opportunity than was the case in the data of the late 1980's.

Now let me turn to the telling data on putting our money were our mouth is -- how much should we be spending on the space program? In general (Figure 13), a majority of people seem to favor investment in the space program; especially when we combine those who favor an increase with those who believe it should be continued at its current levels. Not until the choice between "investment in space or...on domestic programs" do we see a significant switch in 1990 in favor of the domestic programs. I maintain that choice is not a real one. We obviously do not spend the money in space; it is actually spent in this country and it is a benefit to our domestic economy, to our technological development and to our competitiveness and job base. I feel strongly that the space effort is the peaceful alternative to the cutback in our defense effort That may, in fact, turn out to be an effective message and a persuasive one in getting recognition for the importance, benefits and need for such a mission and such a space program. However, determining whether that is the case will require meaningful message research and evaluation.

What are the conclusions that can be drawn from all this attitude research on the space program? Here (Figure 14) are my conclusions. The attitudes concerning the space program are generally favorable, especially when we consider the economic problems our nation faces. However, many of the comments made are in response to suggested goals, benefits, etc. There is very little research that is open-ended and seeks out the level of understanding that the public actually has about the space program and the extent that they actually think about it themselves. We need such greater searching research.

It is significant that there is no research into the attitudes of the public concerning the use of nuclear systems in space nor in determining what they would think about all the nuclear systems that have already been used in space. We need greater understanding of those views.

My next three conclusions all relate to the need for an effective program that can communicate to the public and to policy makers the benefits and importance of and the need for the space program. We must determine what messages are truly effective and then devise a broad array of approaches to communicate those messages to the public and to decision and policy makers. We have no such program now. In fact, I would have expected the President's SEI goals to have become the basis for a comprehensive program planning and communication effort. But I certainly did not see that develop and I do not see it available or being developed to the level required.

Therefore, my major conclusion, punch line and appeal to all those informed on and involved in this country's space program is that we establish a strong, effective communications program that will convey the benefits of the program and rebuild the enthusiasm for space activities we used to have. LET'S GET ON WITH THAT JOB.

FIGURE 1

ATTITUDES TOWARD NUCLEAR ENERGY

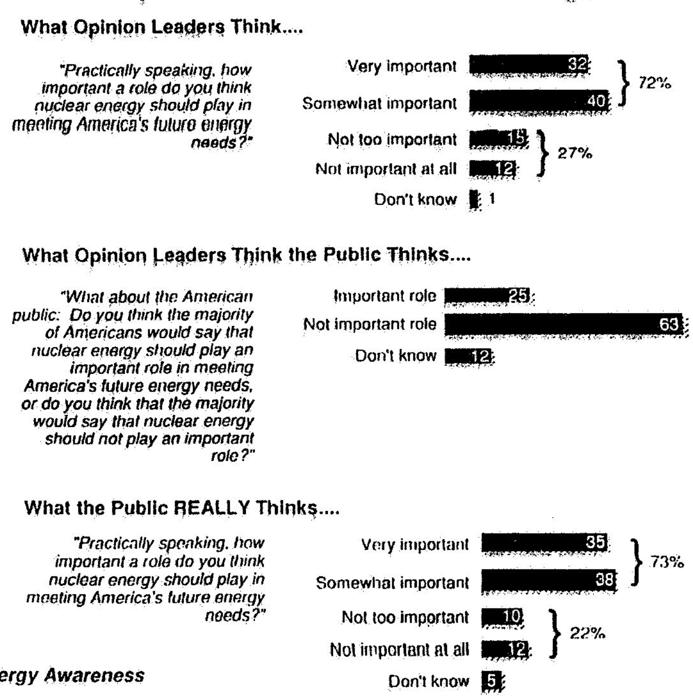
NUCLEAR ENERGY TO PLAY IMPORTANT ROLE	80%
NUCLEAR ENERGY SHOULD PLAY IMPORTANT ROLE	73%
NEED FOR NUCLEAR ENERGY TO INCREASE	76%
CLOSE DOWN NUCLEAR PLANTS	15%

FIGURE 2

Big Perception Gap

Real and Perceived Public Opinion About Nuclear Energy

Opinion leaders and the public both favor nuclear energy....but opinion leaders underestimate public support. The gap between real and perceived public opinion is huge.



Prepared by the U.S. Council for Energy Awareness
April 1992

FIGURE 3

GAINING PUBLIC ACCEPTANCE, APPROVAL, AND SUPPORT FOR USING NUCLEAR SYSTEMS IN SPACE MISSIONS

IT'S TIME TO ORGANIZE A PROGRAM TO DO THAT

FIGURE 4

USCEA Ideas About Nuclear Energy Plants

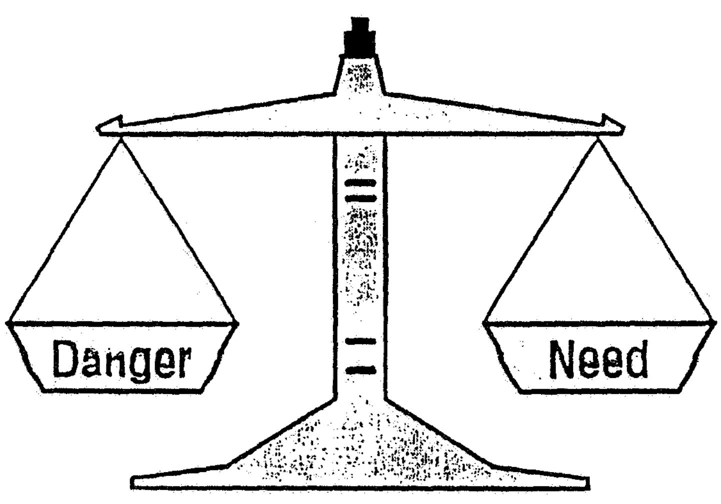


FIGURE 5

USCEA

Ideas About Nuclear Energy Plants

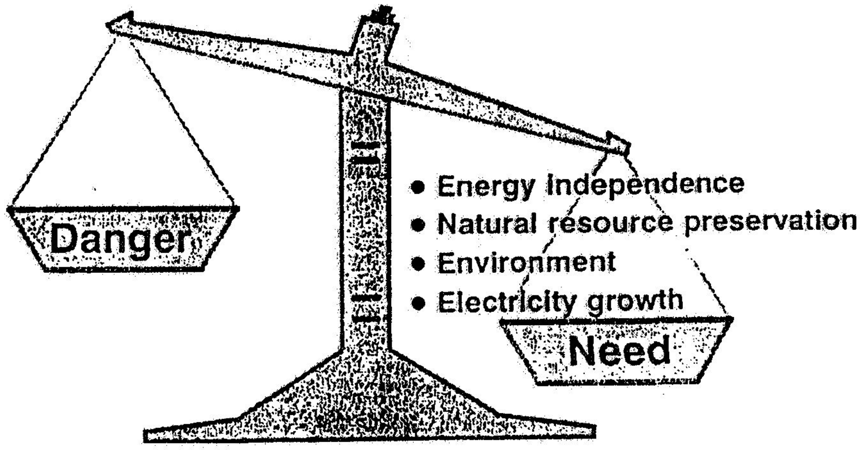


FIGURE 6

GAINING PUBLIC ACCEPTANCE, APPROVAL, AND SUPPORT FOR USING NUCLEAR SYSTEMS IN SPACE MISSIONS

Gaining that acceptance, approval, and support requires first gaining recognition of the need for and the benefits of using those nuclear systems in space.

We do not use nuclear energy in space unless the benefit and need are clear.

THEREFORE, THE OBJECTIVE IS FIRST TO GAIN PUBLIC RECOGNITION, ACCEPTANCE, APPROVAL AND POLITICAL SUPPORT FOR THE SPACE PROGRAM BROADLY; AND FOR MISSIONS THAT BENEFIT SUBSTANTIALLY FROM OR REALISTICALLY REQUIRE NUCLEAR SYSTEMS FOR THEIR ACCOMPLISHMENT.

FIGURE 7

DEVELOPMENT OF AN EFFECTIVE PUBLIC COMMUNICATION PROGRAM REQUIRES A SOLID BASE OF ATTITUDE RESEARCH

- Public attitude tracking
- Strategy and message testing
- Testing communication vehicles
- Evaluation of communication effects

FIGURE 8

ATTITUDES TOWARD SPACE PROGRAM

Support space program overall	80%	(Mar. 90)
Space program is important to U. S.	88%	(June 88)
Approve of America's civilian space program	80%	(July 88 & Feb. 90)
U.S. lead in space technology important	82%	(Feb. 88)

Data provided by Roper Center, University of Connecticut; from Rockwell - Market Opinion Research; and Yankelovich - Time Magazine sources.

FIGURE 9

IMPORTANCE OF THE SPACE PROGRAM

	JULY 1988	FEB. 1990
To our country	88%	82%
To you personally	71%	68%
Space exploration very important to the U.S. and the world	71%	67%
Space exploration is a luxury with all the problems here on Earth	25%	29%
Benefits of space program will be more important 10 years from now*	72%	
Looking back 20 years; time, effort and money to land men on the moon was worth it.	77%	

Data from Rockwell - Market Opinion Research Surveys
Date noted by * from Gordon S. Black Corporation, taken from U.S.A. Today

FIGURE 10

IMPORTANCE OF REASONS FOR SUPPORTING THE U.S. SPACE PROGRAM

	JULY 1983	FEB. 1990	FEB. 1992
Makes possible new and important scientific and medical discoveries	90%	89%	92%
Provides new and improved consumer products and services	76%	76%	74%
Develops new technology to improve U.S. productivity and economic competitiveness	87%	87%	88%
Helps military defend country	80%	79%	80%
New frontier, important to pioneering and exploration heritage	82%	79%	
Space leadership strengthens America's worldwide prestige	81%	69%	
Helps us understand weather, climate, environment		92%	88%
Helps interest young people in science and engineering studies		88%	88%

Data from Rockwell • Market Opinion Research and Yankelovich

FIGURE 11

U.S./NASA SPACE GOALS

	JULY 1988	FEB. 1990	FEB. 1992
Improve understanding of climate, weather, atmosphere - start new satellite and Space Station program with international participation	86%	81%	
Explore solar system with unmanned flights	82%	85%	71%
Permanent manned U.S. Space Station with international participation	78%	74%	65%
Back to the Moon — Base for scientific research and mining lunar materials	70%	64%	57%
Manned mission to Mars — Science outpost and exploration	66%	62%	49%

Data from Rockwell - Market Opinion Research and Yankelovich Surveys

FIGURE 12

ATTITUDES ON MANNED MARS MISSION

1988:	Good idea to cooperate with Soviet Union on Mars Mission	71%
	Yankelovich-Time Survey	
1988:	Increase NASA budget to permit manned Mars mission	64%
	Rockwell Opinion Research	
1988:	If you favor manned Mars mission:	
	Should U.S. go independently?	31%
	or equal partners with Russians?	54%
	Rockwell Opinion Research	
1989:	Where should astronauts go next?	
	Permanent Space Stations?	40%
	Planet Mars?	14%
	Back to the moon?	7%
	Somewhere else?	9%
	Don't send anywhere	20%
	Gordon Black Corporation	

FIGURE 12 (continued)

ATTITUDES ON MANNED MARS MISSION
continued

1989:	What should be the top priority of the Space Program?	
	Basic research - solar system and planets	30%
	Zero-G and commercial technologies	18%
	Space based defense shield	14%
	Mining resources on Moon and planets	23%
	Gallup	
1989:	How important for the U.S. to be first on Mars?	51% vs. 48%
	Gallup	
1991	How important for the U.S. to be first on Mars?	35% vs. 64%
	Gallup	
1990:	Manned missions to Moon and Mars will encourage science and engineering studies	81%
	Rockwell Opinion Research	
1990:	Favor President Bush's SEI missions*	69%
	Rockwell Opinion Research	

*38% of the people are aware; 61% are not aware of SEI proposals

FIGURE 13

AMOUNT OF EFFORT ON THE SPACE PROGRAM

(Rockwell Supported Research)	JULY 1988	FEB. 1990	FEB. 1992
Space program should be expanded	65%	53%	58%
Space program should continue as is	63%	66%	67%
Expenditures should be cut back	36%	40%	42%
U.S. should spend whatever necessary to maintain leadership in space	61%	56%	63%

FIGURE 13 (CONTINUED)

AMOUNT OF EFFORT ON THE SPACE PROGRAM

continued

	JULY 1988	JULY 1990	JULY 1992	JAN. 1990 *
Amount of money being spent on U.S. space program should be:				
Increased	26%	27%	17%	19%
Kept the same	41%	42%	37%	40%
Reduced/eliminated	24%	22%	32%	38%

Gallup Survey (* Marist Inst. Survey)

	JULY 1988	JULY 1990	JULY 1992	JAN. 1990 *
Is investment in space worthwhile or better spent on domestic programs?				
Worthwhile	43%		39%	
Domestic programs	52%		57%	

Gallup Survey

FIGURE 14

CONCLUSIONS

- Generally, favorable attitudes on space program
- Much of the comment was based on suggestions with very little open-ended, volunteered comment
- No data on using nuclear energy in space or on contributions already made by nuclear energy
- No significant, coordinated communications program exists
- No system for communicating with influentials and the public by constituents, scientists, etc.
- No actual message testing to define effective ones
- President Bush's SEI was not grabbed, pushed, nor run with as the basis for building public and political support
- No clear long-term program laid out with clear short and intermediate term milestones as the basis for developing and demonstrating SEI technologies.

FIGURE 14 (CONTINUED)

CONCLUSIONS

CONTINUED

A STRONG, EFFECTIVE COMMUNICATIONS PROGRAM IS REQUIRED TO REBUILD ENTHUSIASM FOR SPACE ACTIVITIES AND TO HOLD IT. THE BENEFITS TO THE NATION AND TO AMERICANS JUSTIFIES IT.

Let's start with one that will feed into the existing communications of various companies, associations, research organizations and government.

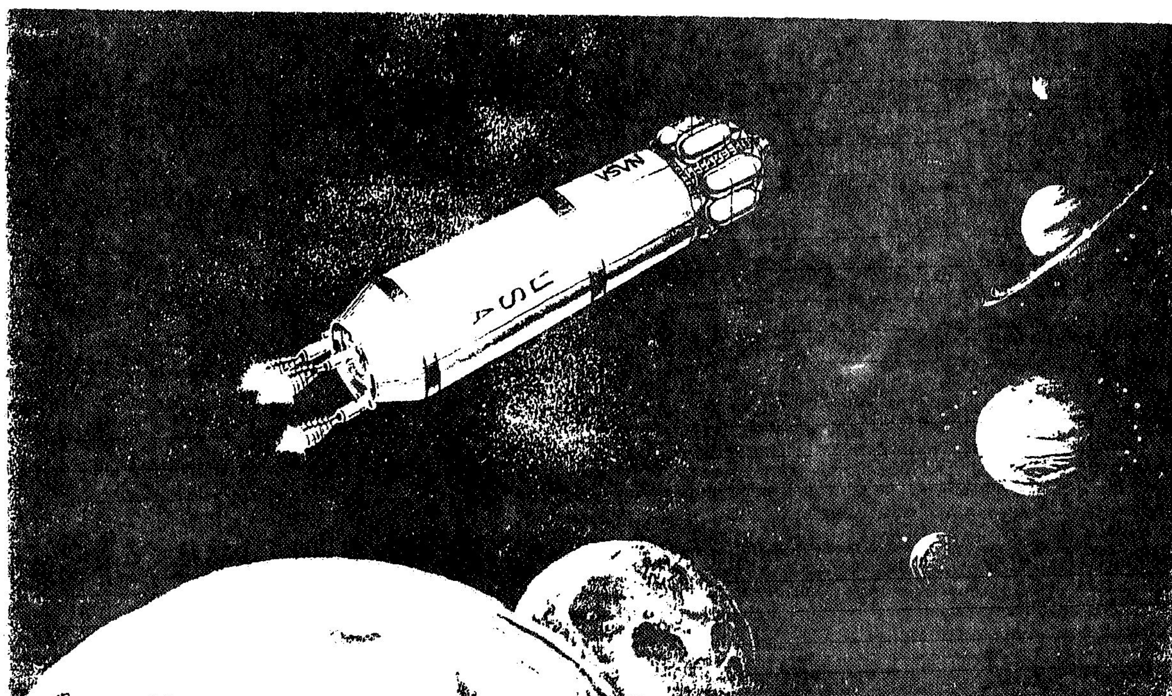
N93-26918

Rover/NERVA-Derived Near-Term Nuclear Propulsion

FY92 Final Review

at

NASA-LeRC
October 22, 1992



Agenda

- Introduction
- Reactor Concept Development
- Engine Conceptual Design
- Key Technology and Streamline Development Plan Assessment

Rockwell International
Rocketdyne Division

Westinghouse Electric Corporation

Introduction

FY92 accomplishments centered on conceptual design and analyses for 25K, 50K, and 75K engines, with emphasis on the 50K engine, to NASA requirements.

During the first period of performance flow and energy balances were prepared for each engine size with single and dual turbopumps. Plan, elevation, and isometric drawings were prepared for each of these configurations, and thrusts-to-weight were estimated. A review of fuel technology and key data from the Rover/NERVA program, established a baseline for proven reactor performance and areas of enhancement to meet near-term goals. Studies were performed of the criticality and temperature profiles for probable fuel and moderator loadings for the three engine sizes, with a more detailed analysis of the 50K size.

During the second period of performance, analyses of the 50K engine continued. A chamber/nozzle contour was selected and heat transfer and fatigue analyses were performed for likely materials of construction. Reactor analyses were performed to determine component radiation heating rates, reactor radiation fields, water immersion poisoning requirements, temperature limits for restartability, and a tie-tube thermal analysis. In addition reactor safety and reliability were assessed.

Finally, a brief assessment of key enabling technologies was made, with a view toward identifying development issues and identification of the critical path toward achieving engine qualification within 10 years. Our initial appraisal suggests that critical path for the program will be the design, construction, and acceptance testing of engine test facilities.

Requirements

- Rover/NERVA-derived technology
- "Near-Term" man-rated mission
- 4.5 hours qualification test at rated conditions to validate 1.5 hours at rated conditions for manned missions
- Restartable, at least 10 starts
- Launch envelope, 30 m (length) x 10 m (diameter)
- I_{sp} > 850 seconds
- Thrust

  A. Initially--25K, 50K, and 75K
  B. Continued effort--50K

- Thrust/Weight (with internal shield) ≥ 4

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Requirements

Requirements for the FY92 NASA-funded effort derive from the Statement of Work. The basic objective was the assessement of the near-term feasibility of Rover/NERVA-derived nuclear thermal rocket engine technology for meeting piloted missions to Mars. The basic requirements for the engine provided by NASA included size limits, target specific impulse, number of restarts, operating life, and thrust-to-weight lower limit. Initial analyses were to be performed for three engine thrust sizes: 25K, 50K, and 75K. Final concept development was to be performed for the 50K thrust size engine.

Additional Ground Rules

- Pewee fuel element, temperature, and ZrH moderator
  - Chamber temperature 2,550 K
  - Power density 1.18 MW/element
- Tie tubes with expander cycle
- Dual turbopumps/loss of both pumps
- Nozzle expansion ratio, 200/1
- Radiation leakage limits from NERVA
- System requirements of NASA N.P. 002

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Additional Ground Rules

Communication with NASA subsequent to issuance of the Statement of Work provided additional guidance: Pewee operating parameters for chamber temperature and power density, use of tie tubes with the expander cycle, incorporation of dual turbopumps with consideration of pump outages, a nozzle expansion area ratio (200), radiation limits from the NERVA design, and additional system requirements found in NASA N.P. 002, "Nuclear Thermal Rocket Engine Requirements."

NERVA-Derived 50K Engine Schematic

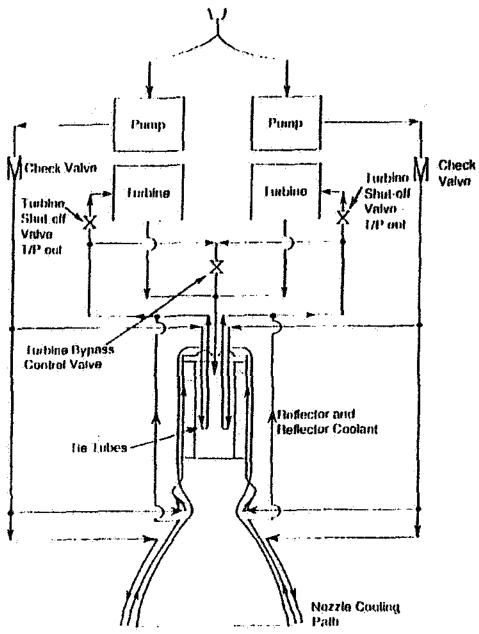
Chamber pressure = 784 psia

Chamber temperature = 2,550 K

Specific impulse = 870 seconds

Nozzle expansion ratio = 200:1

Nozzle bell = 110% length



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NERVA-Derived 50K Engine Schematic

The 50K engine features dual turbopumps supplying liquid hydrogen to the tie tubes, and the chamber and nozzle. Approximately 70% of the flow goes to cool the tie tubes and moderator; the heat pickup provides the energy for the turbines. The propellant flow used to cool the chamber and nozzle also cools the reflector and pressure vessel. The total flow is mixed together, flows through the fuel elements where the temperature is increased to 2,550 K, and is exhausted from the nozzle to produce thrust. The engine is sized and packaged to fit within given geometrical constraints; consequently, chamber pressure and bell nozzle length are selected to maximize specific impulse and thrust-to-weight.

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Key Features/Attributes

- Proven technology, low risk approach
  - Nozzle technology flying with Space Shuttle
  - Existing turbopump designs applicable
  - Rover/NERVA-derived reactor
  - Minimum development time/money
  - Supports 10-year qualification goal
- I_{sp} > 150 seconds better than NERVA-XE'
- MCNP permits fuel loading for flat profile
- Tie-tube support approach facilitates
  - Expander cycle turbine for improved I_{sp}
  - Incorporation of ZrH to minimize reactor size
- Optimized packaging and flow balancing
- Can accept evolutionary improvements

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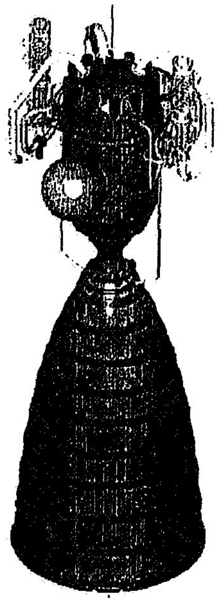
Key Features/Attributes

The Rocketdyne-Westinghouse nuclear thermal rocket engine benefits from a combination of the technology proven in the Rover/NERVA program and modern rocket engine man-rated components. The goal of producing a qualified engine within 10-years can be achieved with minimal development, based on the current state of the art. Cooled chamber and nozzle technology from the SSME is directly applicable, and turbopumps from the J-2S, Rover, and SSME bracket current requirements. Studies were initiated to examine pump-out performance with boost pumps and multiple turbopumps; however, meaningful results were not achieved within the allocable funding limitations.

Easily achievable enhancements provide improvements in I_{sp} over the last NERVA engine tested, NRX-XE'. Incorporation of tie tubes and the expander cycle, increase of the expansion ratio from 10 to 200, regenerative cooling throughout, and increase of the chamber temperature to the Pewee conditions adds over 150 seconds of specific impulse. A further increase of chamber temperature to 2,700 K by use of composite elements would add another 30 seconds to bring the total to 900 seconds.

The preliminary configuration has the turbopumps at the side of the chamber to shorten the overall length of the engine assembly. Within that configuration flow and energy balances are optimized to minimize pressure which directly affects ducting wall thickness.

NERVA-Derived 50K Engine Isometric



Reactor exit temperature = 2,550 K

Dual turbopumps

Split-flow expander cycle

Nozzle expansion ratio = 200:1

Nozzle bell = 110% length

Specific impulse = 870 seconds

Thrust/weight = 5.3 (with shield)

Engine length = 7.6 m

Exit diameter = 2.4 m

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NERVA-Derived 50K Engine Isometric

A key feature of the engine includes compact packaging, with turbopumps mounted to the side of the reactor vessel to reduce the overall height and permit a higher expansion within the geometrical constraints. Another feature has the tubular nozzle attaching to the chamber at a low expansion ratio to save weight and to facilitate ground testing. An area for evolutionary change in this design would be the substitution of uncooled composite ceramic materials for the tubular nozzle for a potential weight saving and some increase in specific impulse.

Technology Assessment Results

- Technology available for most issues

  Rover/NERVA, SSME, Rocketdyne state of the art, SP-100, terrestrial advanced reactors, state-of-the-art electronics and computers

- Unresolved system design issues

  Loss of turbopumps, lifetime, intact reentry--water subcriticality (or total dispersal), decay heat removal, engine-out cooling during operations, fuel midband corrosion

- Critical path is engine test facility

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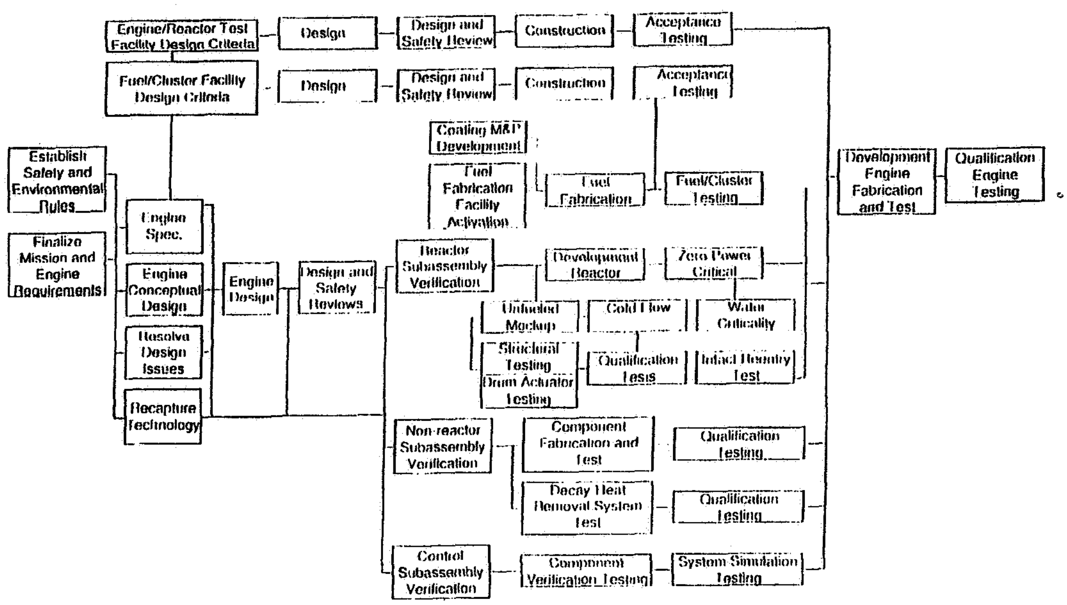
Technology Assessment Results

The assessment of key technologies led to the conclusions that (1) existing technology in reactors and engine systems is applicable to most design areas, (2) there are issues requiring attention early in the program to assure satisfactory resolution, and (3) the assured early availability of an engine/reactor test facility is critical to meet, successfully meet the 10-year engine qualification goal.

NTR Streamline Development Logic



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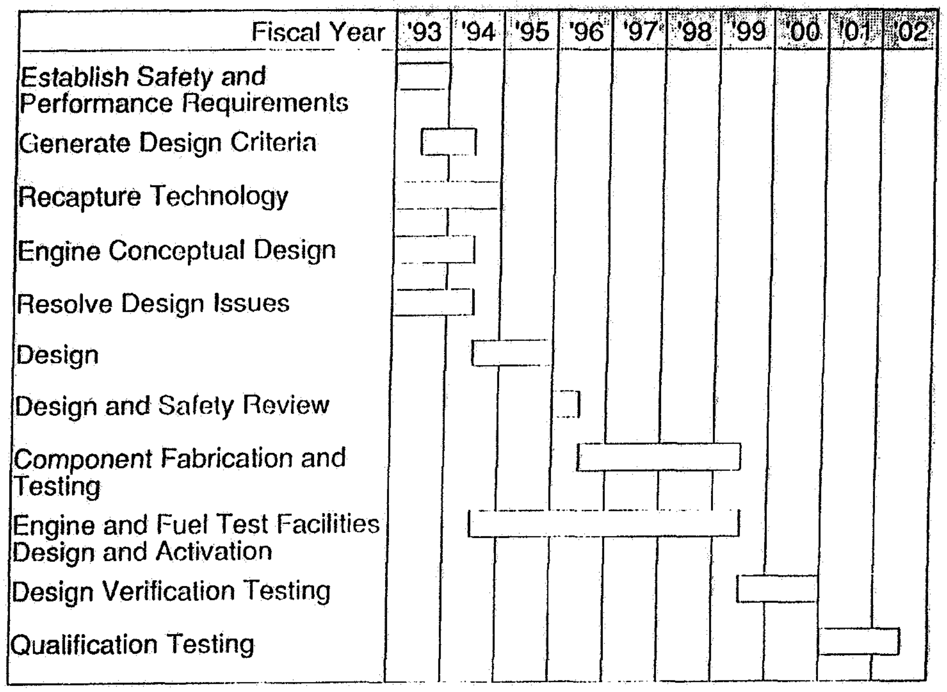
NTR Streamline Development Logic

A development logic diagram can include many layers of detail and be organized in many different ways. This high-level diagram shows many necessary tasks in setting requirements, recapturing technology, resolution of key design issues, facility design, construction, and activation, and testing of components and systems. The most important message is that the program must start with well-defined requirements and design criteria, and that the availability of key test facilities will drive the rate of achievement of the 10-year goals. Near-term activities of conceptual design, technology recovery, and resolution of design issues will provide a sound basis for proceeding quickly as substantial funding becomes available.

NTR Streamline Development Plan Summary



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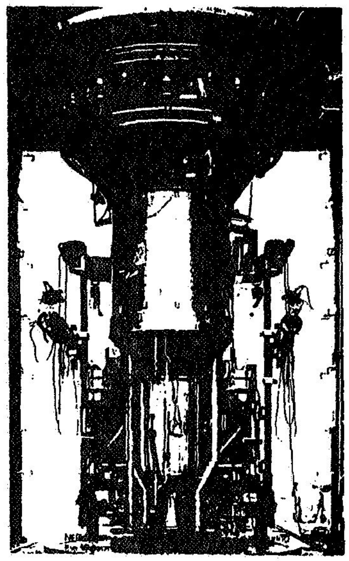
NTR Streamline Development Plan Summary

The time-phasing of key groups of activities from the development logic diagram shows that several tasks should be emphasized at the start: setting requirements, technology recapture, and establishing design criteria. Test facility design, construction and activation must also begin promptly to assure that the 10-year schedule can be met.

REACTOR CONCEPT DEVELOPMENT



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- NERVA Derivative Reactor Concept Design
- NTR Nuclear Parameter Study
- Analysis of Reactor for 50K/lb$_f$ Engine
- Assessment of Fuel Technology
- Assessment of Nuclear Safety Issues
- Summary and Conclusions

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Development of a nuclear thermal rocket design concept for Fast Track studies is based on the NERVA/Rover technology database. Design analyses to provide NTR designs to meet program requirements are developed with current design methodologies benchmarked to NERVA/Rover technology. The NERVA derivative reactor concept design is based on NERVA R-1 reactor design with design features upgraded to include the demonstrated capabilities of the NERVA/Rover program. A historical summary of the completed tests of the NERVA/Rover program and the NTR performance demonstrated by test results are summarized in the following pages.

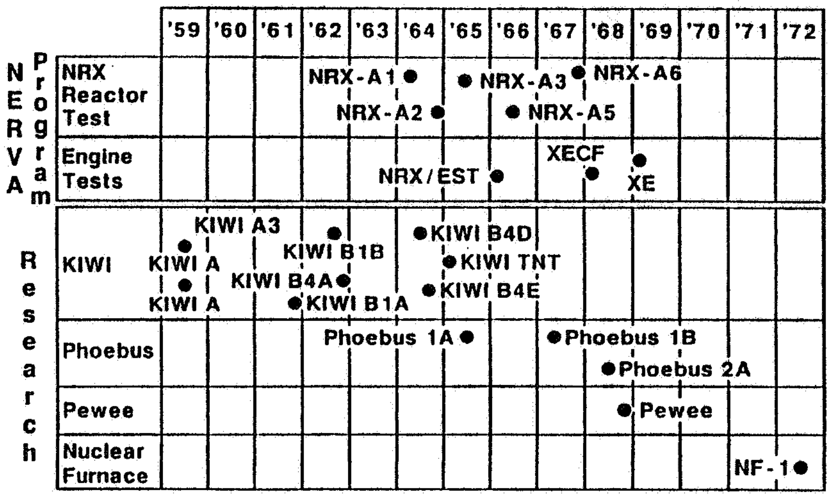
Based on a set of NASA directives, parametric analyses of the size and performance characteristics of NTR reactors which provide performance consistent with 25K, 50K, and 75K lb$_f$ engines was completed. Later discussions show the results of more detailed studies on the reactor design for the 50K lb$_f$ engine.

Based on a review of the NERVA/Rover technology database, a current assessment of the fuel technology and nuclear safety issues for the application of the NERVA derivative reactor in the NTR program is discussed.

In summary, the lessons learned during the conduct of the work tasks are discussed.

NERVA/Rover REACTOR SYSTEM TEST SEQUENCE

The fast track engine draws upon the existing 1.4 billion dollar technology base developed by Los Alamos National Laboratory and Westinghouse during the NERVA/Rover Nuclear Rocket Engine Program.



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The extent of the NERVA/Rover technology is demonstrated by the number of reactor and engine tests completed over the 1959-1972 time frame. The reactor tests completed in the KIWI/PHOEBUS/PEEWEE series demonstrated the wide range in reactor size and power capability provided by the technology. The NERVA test series culminating in the NRX-A6 and XE-Prime tests demonstrated lifetime and performance capabilities of the NERVA/Rover-based NTR's. The NERVA program successfully completed the preliminary design of the R-1 reactor design and the Fast Track reactor designs developed in the current work tasks are derived from the extensive technology database of the NERVA/Rover programs.

Demonstrated Technology

Rover/NERVA Test History

Reactor ID	Chamber Temp. (K)	Fuel Exit Temp. (K)	Space Equiv. ISP (sec)	Time at Full Power (min)	Fuel Type	MWt/ Thrust (kN)
KIWI-B4D	1890-2130	2222	780	1	UO2/Graphite	914/204
KIWI-B4E	1890-2100	2389	820	82.5	UC2/Graphite	914/204
NRX-A2	2090	>2200	775	3.4	UC2/Graphite	1100/245
NRX-A3	2244	>2400	820	16.3	UC2/Graphite	1100/245
PHOEBUS-1A	2366	2478	835	10.5	UC2/Graphite	1340/298
NRX-A4(NRX-EST)	2264-2290	>2400	820	28.6	UC2/Graphite	1100/245
NRX-A5	2280-2333	>2400	820	29.6	UC2/Graphite	1100/245
PHOEBUS-1B	2222-2290	2445	828	30	UC2/Graphite	1340/298
NRX-A6	2300-2405	2556	847	62.7	UC2/Graphite	1100/245
PEWEE-1	1835	2550 2750	845 890	43	UC2/Graphite UC2/Graphite	500/111
XE-PRIME	2278	>2400	820	7.8	UC2/Graphite	1100/245
NF-1	—	2450	830	109	Composite/ Carbide	
PHOEBUS-2A TESTED	2256	2306	805	12.5	UC2/Graphite	4100/913
DESIGN	2500	2550	840		UC2/Graphite	5000/1113

The demonstrated capabilities of NERVA/Rover based NTR's is summarized in the following table. The performance levels reached in each of the key tests completed as shown. As shown, the NERVA/Rover technology provides reactor performance capabilities similar to the requirements of the Fast Track program and later discussions show the capability of NERVA/Rover based design concepts to meet the Fast Track program needs.

NERVA Derivative Reactor Concept Design for 50K lbf Thrust Engine

- Layout drawing
- Solid models

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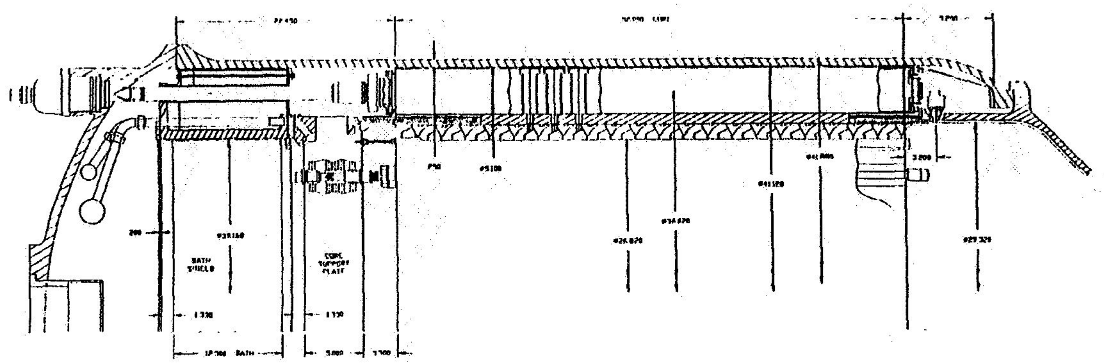

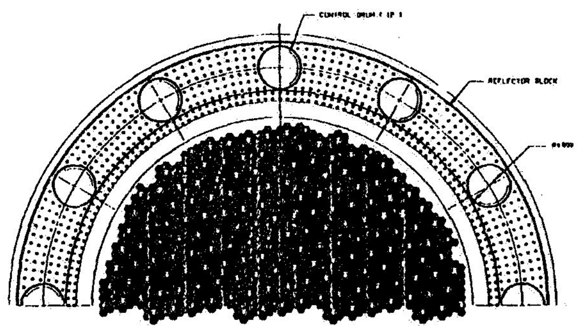

The 50K lb_f engine reactor layout is based on the R-1 NERVA flight reactor design. The R-1 successfully completed an Air Force Preliminary Design Review before the termination of the NERVA project in 1972. The key dimensions of the reactor for the 50 k lbf engine are shown. These were established based on the required engine thrust (core size), and the neutronic requirements (reflector and shield).

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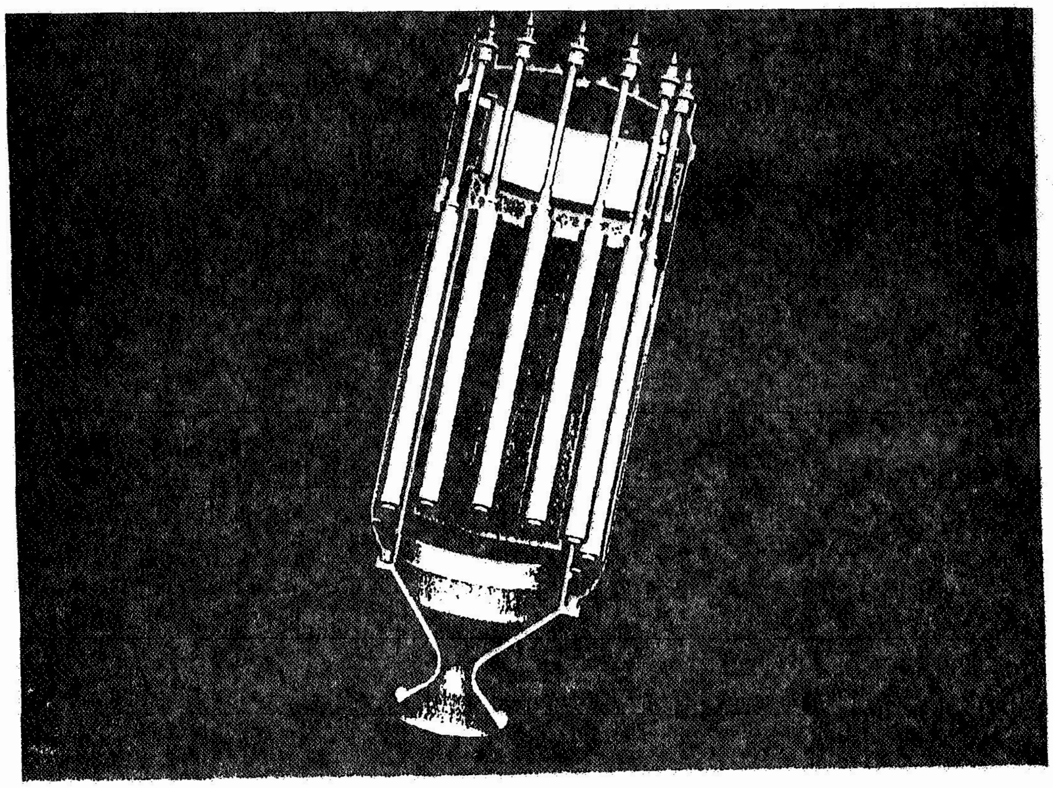
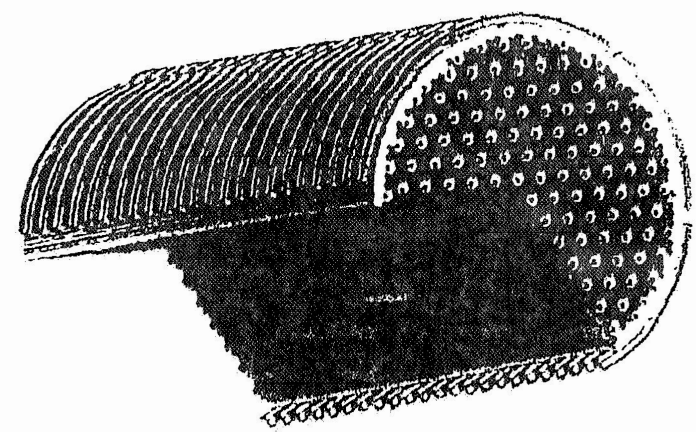


Solid models of the major reactor components and assembly was generated using Pro-Engineer. The viewgraphs show the assembly model of the reactor, where the major components, including the core, can be seen. Since the solid model is parametric in nature, it can be used for a series of reactor sizes of the same type. This makes trade studies easy to perform and weight estimates for these types of reactors can be established fast and accurately. The second major reason for utilizing the Pro-Engineer solid modelling approach is that it provides seamless interface to analytical tools. This integrated approach to design and modelling will be fully utilized in the development of the NERVA derivative reactor design.



NTR Nuclear Parameter Study

- Neutronics Model
- 25K lbf Engine Results
- 50K lbf Engine Results
- 75K lbf Engine Results
- Heterogeneity Evaluation

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Studies of the neutronics design of the NTR were based on three dimensional models derived from the NERVA design. The methodology selected for use in the parametric analyses was the MCNP Monte Carlo radiation transport method. Model parameters of the reactor system were derived from the R-Z model information of the NERVA R-1 reactor system. An automated model generation technique was used to define reactor system models for parametric analyses to size and predict performance characteristics for the various sizes of the NTR system. An R-Z annular ring model of the NTR core configuration was used in parametric analyses in a similar manner to the models in the NERVA database. Three dimensional model details were limited to the reflector control drums and used the geometric modelling capability of the MCNP method. The automated modelling technique and MCNP (Version 3B) were used to define the core and reflector sizes, fuel loading profiles, reactivity worths, and control drum worths and span for three NTR engine sizes; 25, 50, and 75 Klbf thrust levels. In addition, a limited study of the impact of heterogeneous versus homogeneous modelling of the prismatic fuel elements and tie-tubes within the NTR core was performed on a unit cell basis

Study Guidelines

- **NERVA (Prismatic) Fuel Elements**
  52" Long
  0.753" Hex
  19 Coolant Channels
  600 mg/cm^3 Maximum Fuel Loading

- **ZrHx Moderated Tie-Tube**
  SNRC (PeeWee) Maximum ZrHx
  2:1, 3:1, 6:1 Fuel Element to Tie-Tube Ratios

- **Performance**
  1.18 Mw/element
  2550K Chamber Temperature (Point Design)
  784 psia Chamber Pressure

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STUDY GUIDELINES/ASSUMPTIONS

- Reactor Sizes 25K, 50K and 75K lbf Thrust Engines
- Critical Drum Angle of 80°
- NERVA/R-1 Reactor Design Configuration
- R-Z Geometry with Explicit Control Drums
- Neutronics Calculations: MCNP-3B

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Technology Base

Design Features	Reactors						
	NERVA			PeeWee	25K	50K	75K
	Xe-Prime	A-6	R-1				
Tie-Tubes	No	No	Yes	Yes	Yes	Yes	Yes
Ratio, Fe/TT			6:1	3:1	3:1	6:1	6:1
ZrH Loading (Relative)			0.0	1.0	1.0	1.0	0.4
Power, Mwt					512	1024	1536
Core Diameter, In.			35.0		18.8	25.2	30.7
Power Density, MW/FE	0.67	0.67	0.75	1.18	1.18	1.18	1.18
Internal Shield	A1	A1	BATH	--	BATH	BATH	BATH
Fuel Type	Graphite	Graphite	Composite	Graphite	*	*	*

*Not Determined

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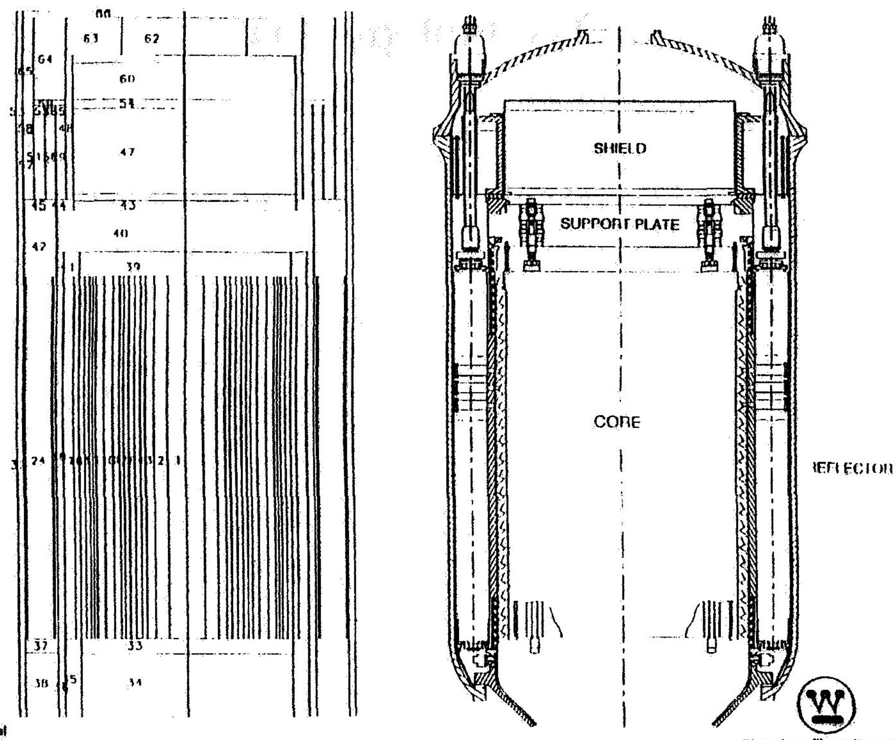
Neutronics Model



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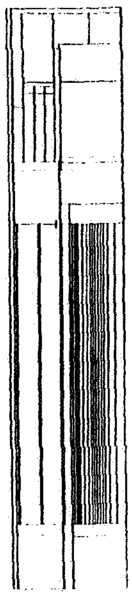
The neutronics model for the NTR system was derived from modelling information in the NERVA database and is shown as an elevation view to illustrate the modelling detail of MCNP models. The MCNP analyses used the ENDF/B-V nuclear data library and were performed in the coupled neutron and photon solution mode to predict region power and required fuel loading to meet target objectives for key neutronics parameters. An actual NERVA system design configuration drawing is depicted to illustrate the modelling approach used.

MCNP Model for 25 Klbf Thrust Reactor



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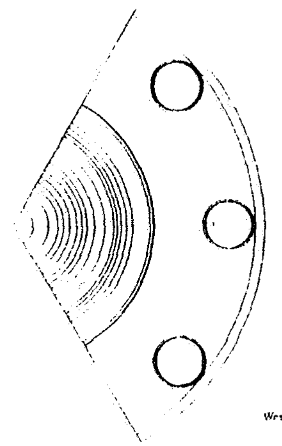
Key Parameters	25K
Effective Core Diamater (in.)	18.80
Fuel Support Ratio	3:1
Core Length (in.)	52.00
Reflector Thickness (in.)	8.00
No. of Drums	9
Peak Fuel Loading	600
ZrH Loading w.r.t. SNRE	1.0



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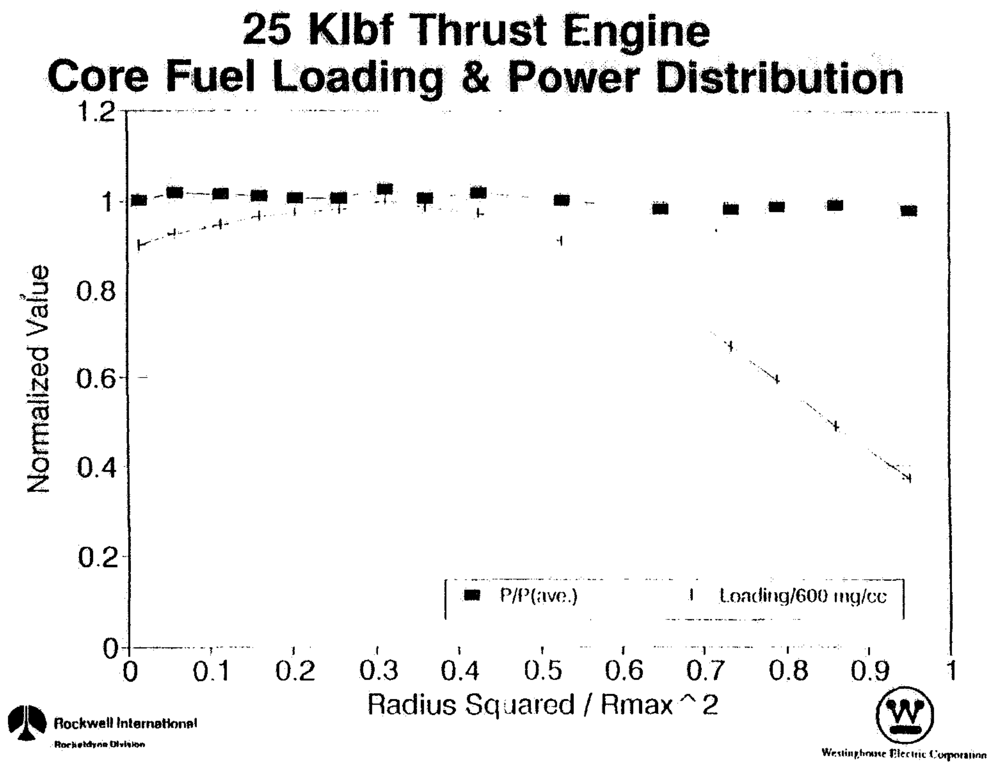
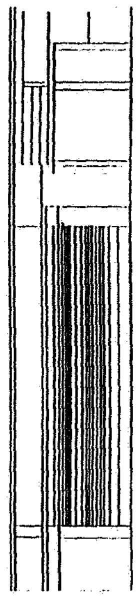
The MCNP model for the 25 Klbf NTR engine and the predicted key parameters are shown in the table on the right. The design bases selected for the small NTR engine size were derived from PEEWEE engine design information with a fuel-to-support tie tube ratio of 3:1, a 52 inch high active core, and 9 control drums of a fixed diameter located a the outer periphery of the Be reflector region. The peak fuel element uranium loading was limited to 600 milligrams/cm^3 and a maximum ZrH loading in the tie-tubes. An iterative process based on MCNP was used to size the reactor core and predict the fuel loading profile to meet the target objectives of an excess reactivity of 0.05 and a flat radial power distribution.

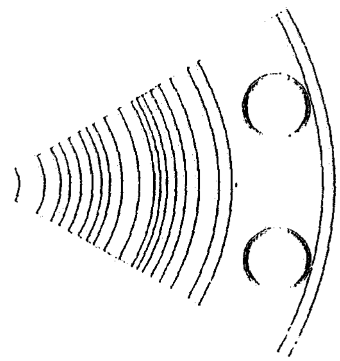


A normalized fuel loading profile predicted for the 25 Klbf NTR engine is shown as a function of normalized area parameter, R^2. The normalized radial power distribution as predicted in the final iteration of the analysis is shown to illustrate convergence to the target objective of a flat or uniform power profile. The MCNP tally method provides the cell or ring average value and more detailed tallying techniques would be required to predict the variation within each fuel annulus.

MCNP Model for 50 Llbf Thrust Reactor



Key Parameters	50K
Effective Core Diamater (in.)	25.18
Fuel:Support Ratio	6:1
Core Length (in.)	52.00
Reflector Thickness (in.)	5.10
No. of Drums	12
Peak Fuel Loading	600
ZrH Loading w.r.t. SNRE	1.0



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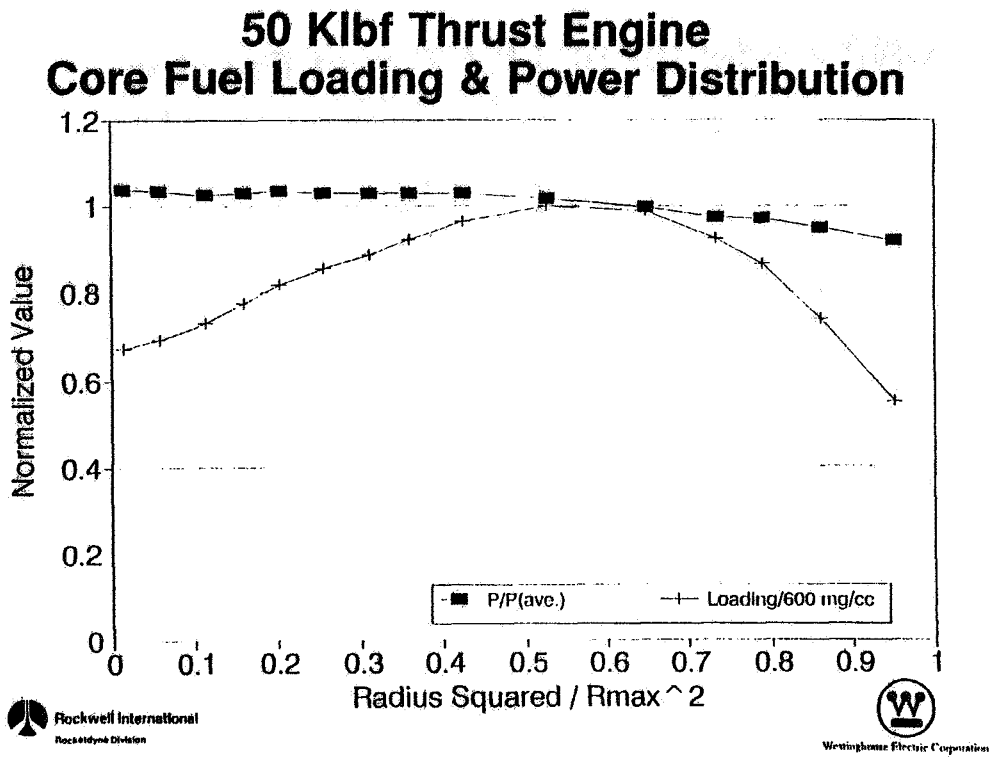
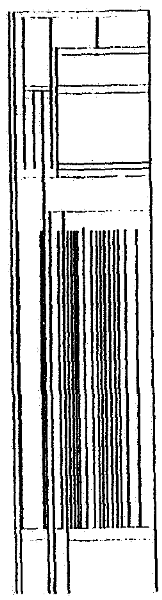
Predicted neutronics parameters for a 50 Klbf NTR engine are shown in the table. Key differences in the design bases selected for this size of engine were a fuel-to-support tie-tube ratio of 6:1 and reflector thickness and number of control drums. The effective core diameter required to meet target objectives is 25.18 inches.

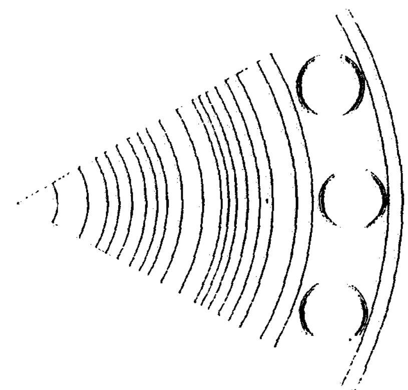


The normalized fuel loading profile predicted for the 50 Klbf NTR engine is shown as a function of normalized area parameter, R^2. As shown, the fuel loading profile differs from the 25 Klbf engine data due to the larger size and the change to a 6:1 fuel:support tie-tube ratio. The lower fuel loading required in the center of the core is related to the change in the moderation of the core and the increase in median fission energy and the effect of radial leakage.

MCNP Model for 75 Klbf Thrust Reactor



Key Parameters	75K
Effective Core Diameter (in.)	30.66
Fuel:Support Ratio	6:1
Core Length (in.)	52.00
Reflector Thickness (in.)	4.785
No. of Drums	18
Peak Fuel Loading	600
ZrH Loading w.r.t. SNRE	0.4



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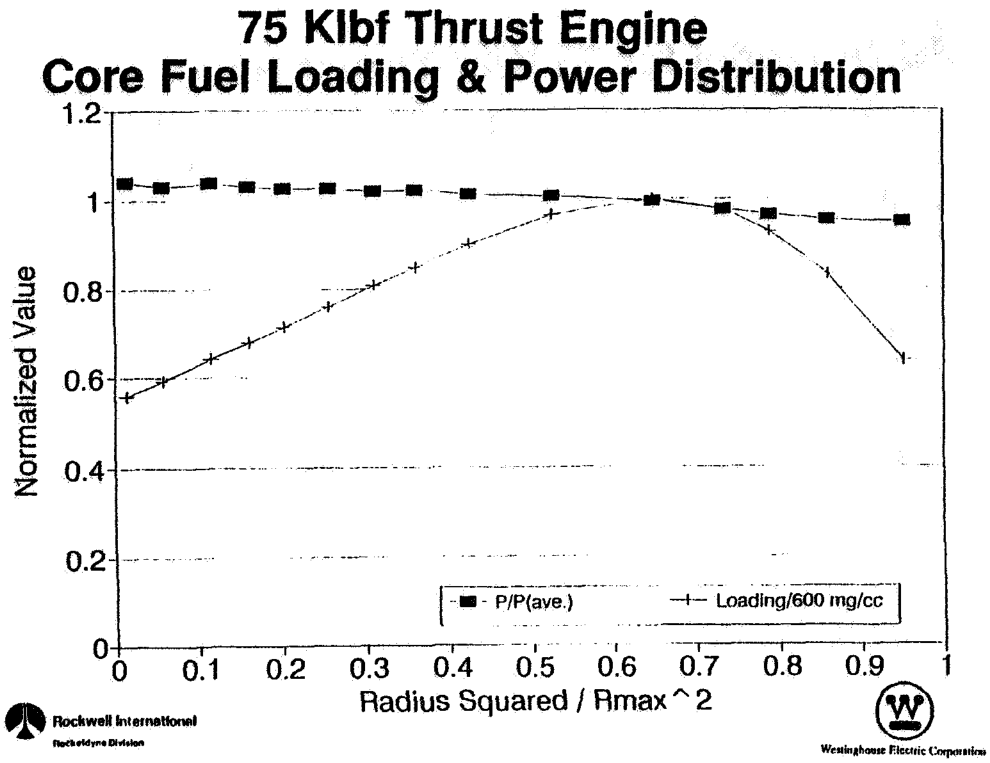
Predicted neutronics parameters for a 75 Klbf NTR engine are shown in the table on the right. The 75 Klbf engine size is similar to the NRX-A6 or R-1 size and the predicted parameters are comparable to the NERVA data. Key differences in the design bases selected for this size of engine were a decrease in the ZrH loading in the support tie-tubes of 0.4 with respect to the SNRE loading. The reflector thickness and number of control drums for the 75K engine are the R-1 dimensions. The effective core diameter required to meet target objectives is 30.66 inches which is similar to the NERVA design..



The normalized fuel loading profile predicted for the 75 Klbf NTR engine is shown as a function of normalized area parameter, R^2. As shown, the fuel loading profile is similar to the 50K engine data and is comparable to NERVA loading profiles.

	Reactor Size, Klbf		
Parameter	25K	50K	75K
Thrust (lb)	25,000	50,000	75,000
Fuel: Support Ratio	3:1	6:1	6:1
Power (MW)	512	1024	1536
Flow (lb/sec)	28.2	56.3	84.5
Core Diameter (in)	18.8	25.2	30.7
ZrH Loading (Relative)	1.0	1.0	0.4
Reflector Thickness (in)	8.0	5.1	4.8
Pressure Vessel OD (in)	38.8	41.9	47.7
Reactor Mass w/o Shield (lb)	5180	6250	8040
Reactor Mass w/Shield (lb)	6590	8080	10480

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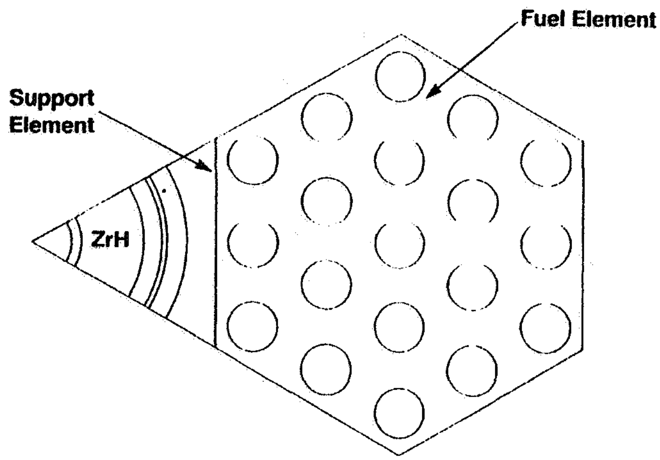
A summary of the results of the preliminary sizing of NTR engines in the 25Klbf-to-75Klbf size range is shown in the table. The design bases used in the parametric analyses are listed on the left. The prismatic fuel element length of 52 inches was adapted from NERVA and fuel performance limits defined based on the PEEWEE data. The predicted masses for the reactor system without and without shielding illustrate the effect of engine size on the engine performance characteristics and sizes. The use of ZrH in the 75K engine size differs from the NERVA design and the impact on a reduced reactor size and mass is shown. The shield masses included in the summary table are based on the same thickness of shield with the mass differences only showing the change in shield diameter.

Heterogeneity Analysis



Unit Cell 6:1 Arrangement

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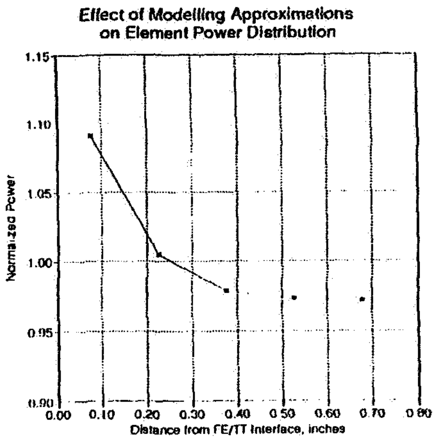
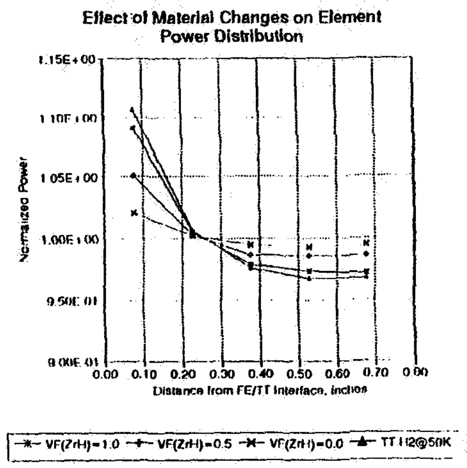
A limited study of the homogeneous region modelling technique for the prismatic fuel element core lattice with ZrH moderated support tie-tubes was carried out using the MCNP method. A unit cell model of a 6:1 fuel-to-support tie-tube configuration includes an annular model of the ZrH moderated tie-tube and the 19 coolant hole prismatic fuel element. A series of unit cell MCNP calculations were run to predict the effect of the ZrH tie-tube on local power distributions and to predict material or material interchange reactivity worths on a unit cell basis.

Effect of Modelling on Element Power Distribution





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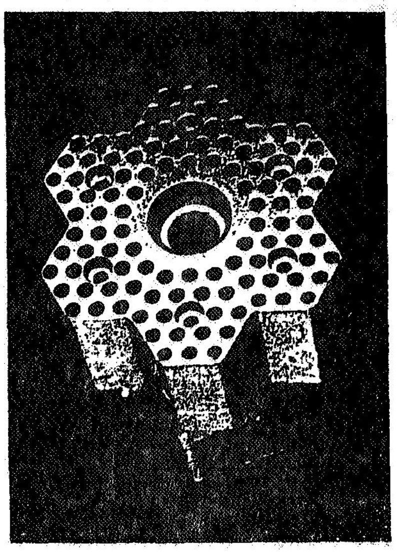
Comparisons of the effect of heterogeneous versus homogeneous modelling on the power distribution in the prismatic fuel assembly is shown in the left figure. The homogeneous model in a unit cell was derived by volume weighting of the prismatic fuel element, tie-tube materials, and hydrogen coolant of the tie-tube and fuel element. The comparison shows a peak to average local channel power of 9-10% for the explicit model of the unit cell. The smear modelling of each fuel element or tie-tube provides similar peak-to-average values. Shown in the right figure is the effect of a decrease in ZrH volume fraction or the introduction of cold (50K) hydrogen in the upward pass of the tie-tube. The maximum effect on local power occurs when the ZrH tie-tube is flooded with H_2 coolant at 50K.

ANALYSES OF 50K ENGINE DESIGN CONCEPT



- Reactivity Coefficients
- Component Nuclear Heating Rates
- Reactor Radiation Fields
  - Shielded (R-1)
  - Unshielded
  - Reduced Shield
- Material Temperature Limit Assessment
- Tie-Tube Thermal Analysis

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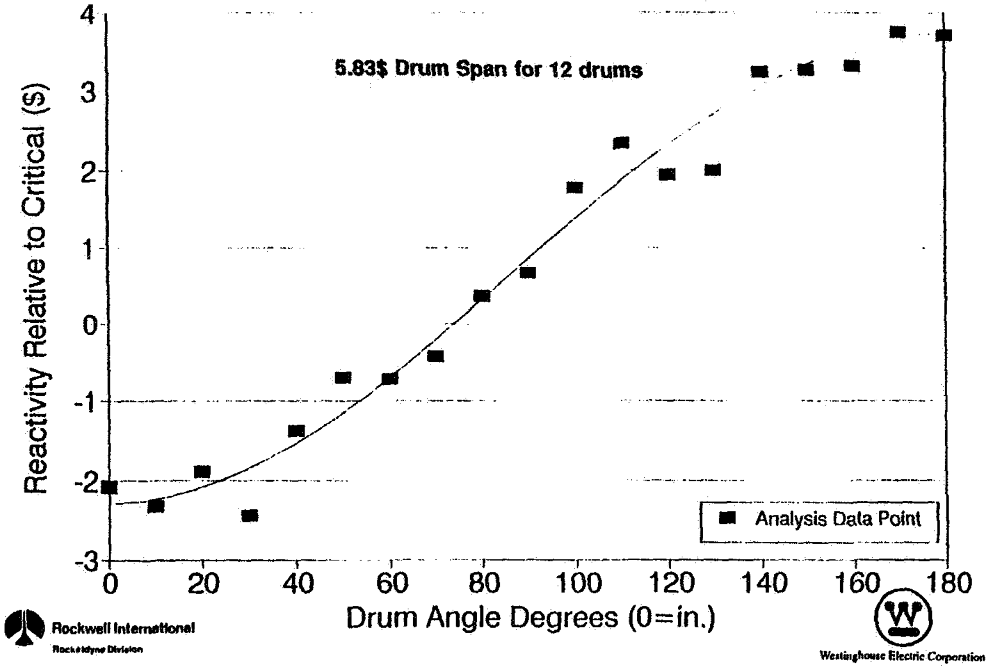
Neutronics analyses of the NTR 50K engine configuration defined earlier were expanded to provide more detailed core performance data. The limited analyses were performed with a more detailed MCNP model to predict the design data for key design parameters as listed on the facing page. Included in the more detailed analyses was; 1) the prediction of reflector control drum worths and span, and 2) reactivity change due to water immersion of the nuclear system. In addition, component nuclear heating rates and radiation fields external to the reactor system are predicted and shown in later pages.

In addition, evaluations of the component temperature limits needed for restartability studies and analyis of tie-tube thermal performance are shown in later pages.

50 Klbf Thrust Engine
Control Drum Reactivity



The predicted reflector control drum reactivity relative to the critical condition is shown on the facing page. The results of the individual MCNP calculations with the explicit modeling of the control drums in MCNP method provide results in agreement with NERVA predictions and illustrate the drum span available for control and shutdown of the 50K engine.

Reactivity Coefficients

Case: 50 Klb$_f$ Thrust Engine

PARAMETER CHANGED	REACTIVITY CHANGED
Drum Worth (@ 80°)	7.3¢/° Rotation
Core Volume	38.9¢/%
Fuel Loading	15.6¢/%
ZrH Loading	19.2¢/%
Reflector Thickness	18.7¢/%
18 Drums (7.34$ span vs. 5.83$ for 12 Drums)	-$2.1

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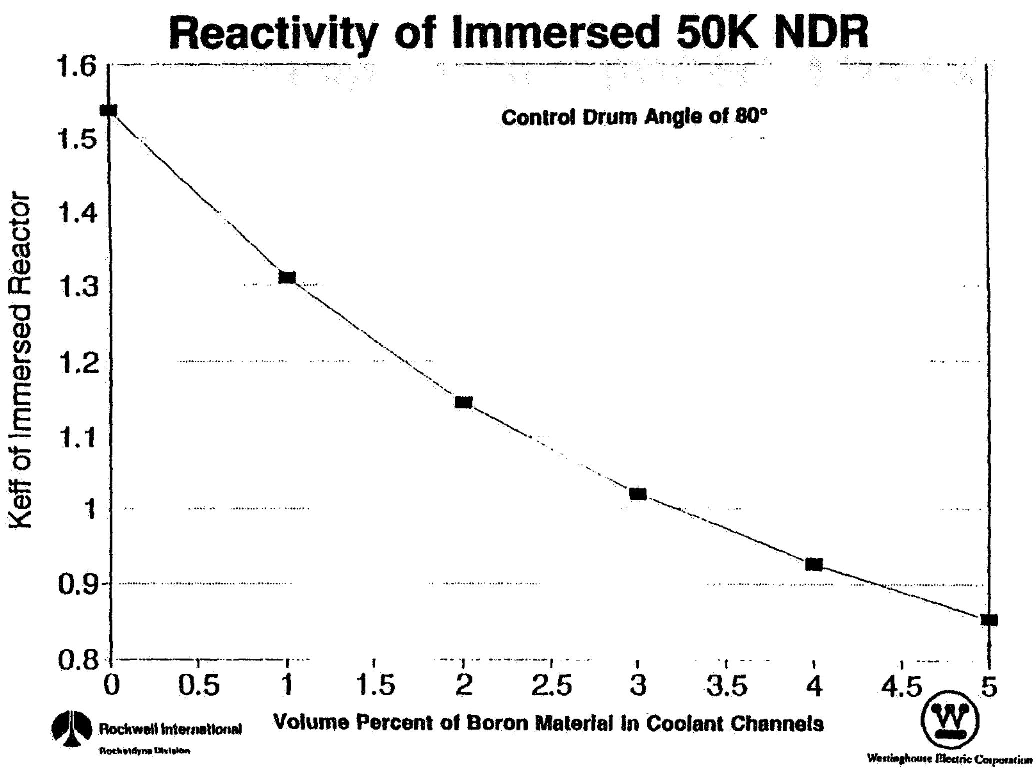
The predicted reactivity coefficients or worths for key design parameters are listed on the facing table. The predicted drum worth is based on the 80 degree postion. The value of 7.3 cents/degree is in close agreement with the NERVA predicted value. Reactivity coefficients for changes in the reactor configuration, fuel loading, ZrH loading in the tie-tubes, and reflector thickness provide data for evaluating design configuration changes. The largest value is the core volume coefficient which is attributed to the change in neutron leakage from the core. Shown also is the effect of changing the number of reflector control drums from 12 to 18 drums.



Predictions of the effect of water immersion of the entire reactor system was modelled in MCNP by replacing the H_2 coolant modelled in each region with water and surrounding the entire system with water. The reflector control drums were parked and a boron-containing material was substituted for a fraction of the fuel element coolant channel volume. The reactivity change from the base case is shown as a function of the volume percent of coolant channel displaced by the boron-containing material. A value of five (5) percent by volume of the coolant channel is a 62 mil boron wire in 7 out of 19 coolant channels in each prismatic fuel element of the core. The reactivity insertion provided by the 5% by volume of boron wires is approximately -74$ with the water immersion of the system resulting in a +50$ reactivity insertion.

COMPONENT HEATING IN 50K REACTOR

COMPONENT	Power, Mw	PERCENT OF TOTAL
Core Fuel and Supports	1000	97.66
Core Periphery (Filler & Seals)	6.6	0.65
Core Barrel Structure	2.4	0.24
Reflector & Control Drums	11.4	1.12
Core Support Plate & Hardware	1.4	0.13
BATH Shield	0.7	0.07
Balance of Reactor	1.4	0.13
Total	1024	

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A summary of the nuclear heating of the major components of the 50K engine is shown on the facing page. The MCNP cell tally method was used to predict the component heating rates.

REACTOR RADIATION FIELD TALLIES IN MCNP-3B CALCULATIONS

(Type and Units)

Radiation Field Type	Energy Bin	Units
Heating Rate In Hydrogen		W/kg
Heating Rate In Carbon		W/kg
Heating Rate In Stainless Steel		W/kg
Neutron Flux		n/cm^2-sec
Neutron Flux, 1 MeV Equivalent In Si		n/cm^2-sec
Neutron Fast Flux	> 1 MeV	n/cm^2-sec
Neutron Intermediate Flux	0.1 MeV - 1.0 MeV	n/cm^2-sec
Neutron Epithermal Flux	0.4eV - 0.1 MeV	n/cm^2-sec
Neutron Thermal Flux	< 0.4eV	n/cm^2-sec
Neutron Dose Rate In Hydrogen		Rad/hr
Neutron Dose Rate In Carbon		Rad/hr
Neutron Dose Rate In Stainless Steel		Rad/hr
Gamma Dose Rate In Hydrogen		Rad/hr
Gamma Dose Rate In Carbon		Rad/hr
Gamma Dose Rate In Stainless Steel		Rad/hr
Gamma Dose Rate In Silicon		Rad/hr

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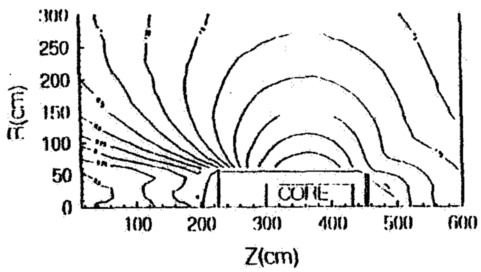
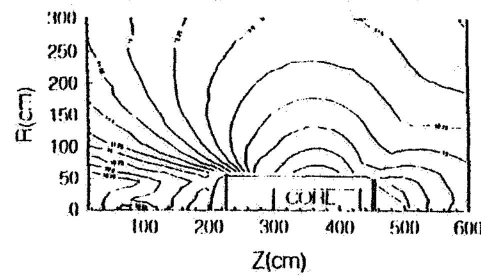
The prediction of the radiation environment external to the 50K engine were performed using the MCNP cell tally methods. Three engine models were analyzed; 1) the conceptual design sized using MCNP in the neutronics design tasks described earlier, 2) all internal shield materials removed, and 3) a modified design with a reduced mass of internal shielding. Each of these models only include the reactor system and the engine components external to the reactor vessel, e.g., tanks, piping, nozzle, are not included in the model. The engine components external to the reactor vessel can contribute to the environment within the internal shield shadow cone and should be included in future studies. The MCNP modelling used a series of annular ring cells imposed external to the MCNP R-Z model of the NTR reactor system for purposes of tallying the desired radiation environments. The facing page summarizes the type of radiation field tallies used in MCNP and either the neutron energy range of the neutron flux tally or the units of heating or neutron or gamma dose rates.

RADIATION FIELDS FOR A 50K SHIELDED REACTOR

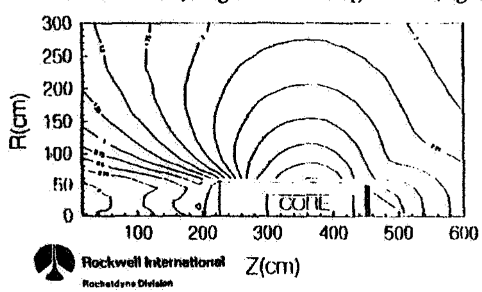
50K Shielded, Log Gamma Rad(C)/hr



50K Shielded, Log Fast Neutron Flux n/cm2-sec



50K Shielded, Log Total Heating Rate W/kg C



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- **Standard R-1 Shield**
- **12.3" Thick Bath Shield**
- **1.3" Thick Lead Shield**

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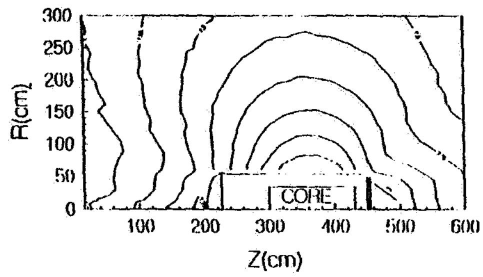
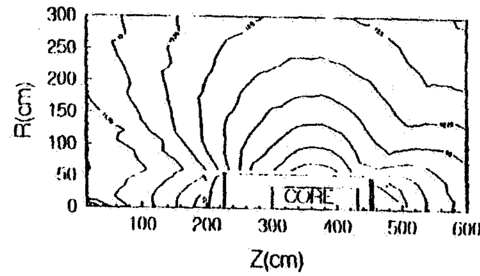
The radiation environment of the original 50K engine design is shown on the next two facing pages for three key tallies. The first 50K engine design used for this analysis included the standard NERVA R-1 internal shield configuration of 12.3 inches (31.25 cm) of BATH shield material and 1.3 inches (3.3 cm) of lead (Pb) shielding. The second page is for an engine design with the internal shields removed. The predicted radiation environments for the shielded case are lower than the design requirements.

RADIATION FIELDS FOR A 50K UNSHIELDED REACTOR

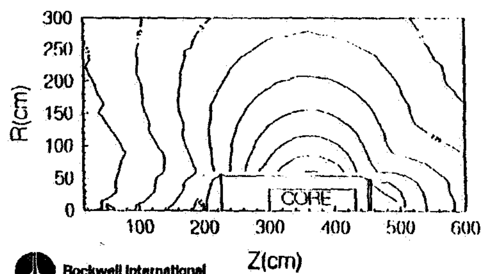
Log Gamma Rad(C)/hr



Log Fast Neutron Flux n/cm2-sec



Log Total Heating Rate W/kg C



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Zero Added Shielding

No Bath or Lead

No Shield Support Plates

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RADIATION FIELDS FOR A 50K REACTOR WITH A REDUCED INTERNAL SHIELD

- Radiation Field Criteria* (in Shield Shadow Cone)
  - Gamma Dose $< 1.8 \times 10^7$ Rad(C)/hr
  - Fast Neutron Flux $< 2.0 \times 10^{12}$ n/cm^2-sec
  - Intermediate Neutron Flux $< 3.0 \times 10^{12}$ n/cm^2-sec
  - Thermal Neutron Flux $< 6.0 \times 10^{11}$ n/cm^2-sec
- Reduced Shield Concept:
  - Eliminates Lead Gamma Shield
  - Reduces BATH Thickness from 12.3" to 9"
- Reduced Shield Peformance ~ 900 lb. Reactor Weight over Standard Shield
  - 900 lb Mass Savings versus R-1 Type
  - Meets Above Criteria (Design Margin > 2.0)

* Per NASA Directive

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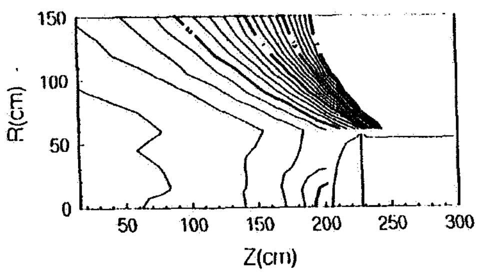
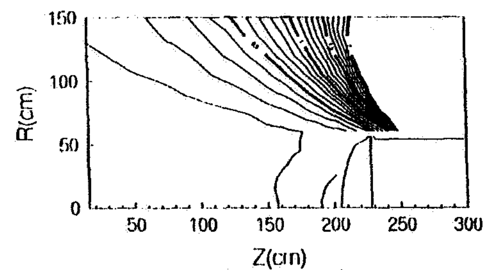
Based on the design requirements imposed on the internal shield design of the NTR engine, a reduced internal shield with nine (9) inches of BATH shield material and no lead (Pb) shielding was modelled and the resulting radiation environments compared to the standard design described earlier. The facing page lists the radiation field design requirements specified for the NTR engine. The reduced shielding configuration meets design requirements with a design margin in the shadow cone of the internal shield of a factor of 2. The design change results in a reduction in shield mass of approximately 900 pounds..

RADIATION FIELD COMPARISON TO CRITERIA FOR 50K WITH REDUCED INTERNAL SHIELD

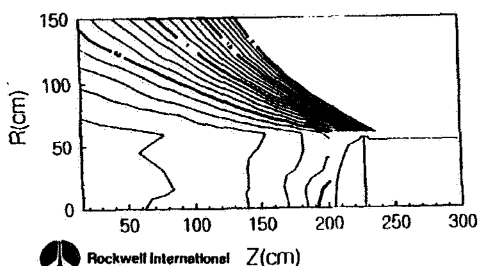
Ratio Gamma Rad/Allowable



Ratio Fast n Flux/Allowable



MAX(Each Ratio Flux/Allowable)



- Shield Size by Allowable Field
- 9.0" Thick Bath, No Lead
- Meets Criteria by a Factor of 2
- Saves 900 lb. in Engine Weight

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The contour plots on the facing page provide data on the performance of the modified shield configuration for the 50K engine relative to the design requirements. The contour data is the ratio of the predicted radiation environment level to the design requirement discussed before. As shown by the data, the reduced shield configuration meets the design requirements within the shadow cone of the internal shield. The design margin in the shadow cone is a factor of 2 or greater in the shadow cone. As discussed before the mass savings of the reduced internal shield design is 900 pounds.

MATERIAL TEMPERATURE LIMITS

REGION	MATERIAL	REUSE TEMP (K)	ALTERNATE MATERIAL**	TEMP (K)
Fuel Element	Graphite	2500		
Other Core Materials	ZrH_x	1000*	No	
	I-718	900	HD-Moly	2000
	A-286	900	Superalloys	~1400
	SS-304	750	Superalloys	~1400
Reflected Materials	Cu-B	1200		
	Be	1400	No	
Vessel Materials	Al-6061	400	Ni, Fe Alloys	
	Ti	800	Ni, Fe Alloys	
Shield Materials	BATH	550		
	Lead	~550	Tungsten	

*Must be pressurized with hydrogen (> 10 TORR)

**No materials identified which provide a capability without significant mass, performance or design penalty

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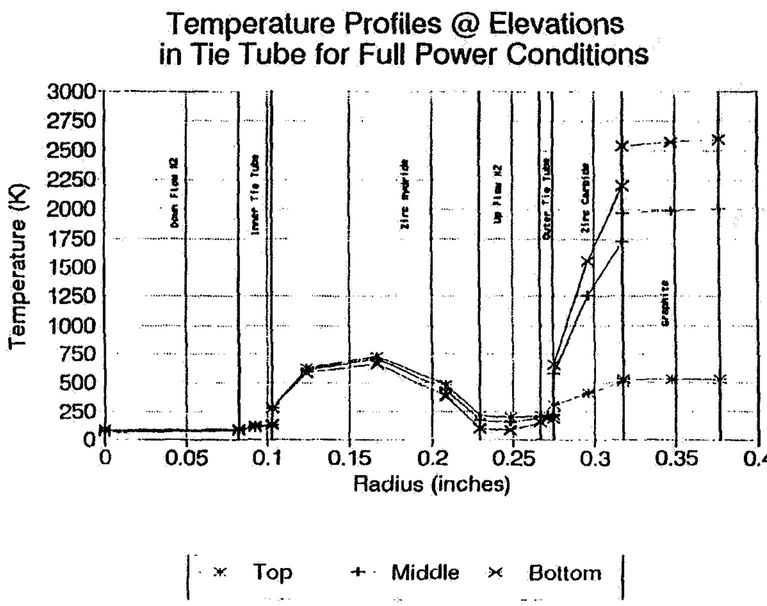

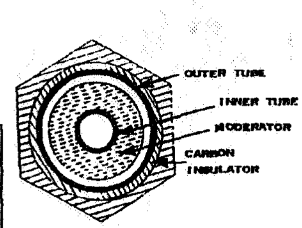

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The tie tube assembly serves two purposes: provides the lateral support for the fuel elements, and heats hydrogen propellant used to drive the turbopump.

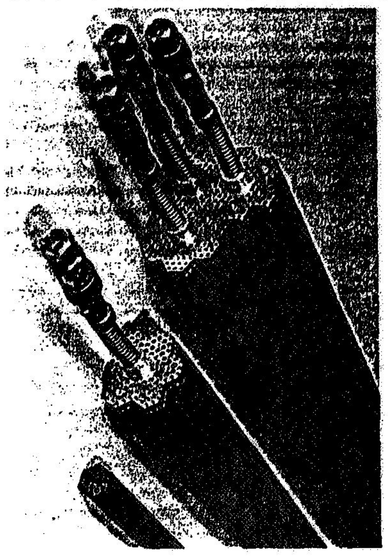
The thermal analysis of the tie tube assembly was performed to establish the adequacy of the design in terms of component temperature and to determine the energy transferred to the hydrogen. The thermal model used for the analysis employed axially dependent heat generation and boundary temperature conditions, temperature and flow dependent hydrogen heat transfer coefficient, and temperature dependent material properties. The thermal model will be used to perform parametric steady-state analysis, as well as transient analysis of throttling conditions.

The radial temperature distribution at three locations (top, middle, and bottom) of the tube assembly is shown on the facing chart for full power conditions.

The temperatures of the ZrH are critical since it has the lowest temperature capability of the materials used in the tie tube assembly. As shown, the maximum calculated temperature for the conditions used exceed 1000 K by a small amount at an internal node in the ZrH cylinder. The calculated heat transferred to the tie tube is 0.18 MW.

The thermal model has been verified against the small engine in the Nuclear Engine Definition Study. The analysis demonstrates that the thermal conductivity of the ZrC insulation is the largest factor in achieving the goal of 0.31 MW per tie tube.

ASSESSMENT OF FUEL TECHNOLOGY



- Review of ROVER/NERVA Test Experience
- Evaluate the Corrosion Mechanisms Affecting Fuel Performance
- Define Problem Areas Needing Near-Term Solution
- Establish Near-Term Fuel Performance Limits
- Compare Near-Term Performance to Fast Track Needs

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An assessment of the NERVA/Rover fuel technology of 1972 is needed to establish expected performance parameters for the Fast Track engine. The fuel life for the Nerva graphite type prismatic fuel element is determined by the amount of graphite weight loss which can be tolerated before the neutronic margin has been lost. The weight loss from the fuel element is due to the corrosive effect of hydrogen on the graphite, which is categorized as either "mid-band corrosion," basically results in a chemical reaction of hydrogen and carbon in intimate contact, or "hot end corrosion," carbon diffusion through a protective coating on the graphite surface.

Great strides were made near the end of the NERVA/Rover program in understanding and eliminating the mid-band corrosion, and it is a basic premise that this corrosion mechanism be suppressed in order to support the needs of the Fast Track program.

Based on the reactor/engine testing program, and the non-nuclear corrosion testing of fuel elements using the improved GEM coatings, the performance limits of "near term" fuel elements were established. The expected fuel element performance was then compared against the needs of the Fast Track program.

Fuel Elements

- **Sustain controlled nuclear heat generation**

  Pyrocarbon coated UC_2 fuel beads dispersed through AXM graphite matrix (630 mg/cc maximum fuel loading

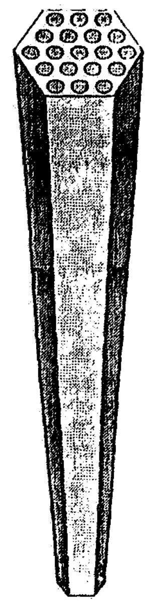
  UC_2-ZrC in composite with graphite (700 mg/cc maximum fuel loading)

- **Limit total reactivity loss to $1.00 at end of life**

  Carbide coating of flow channels

- **Promote heat transfer from fuel element to H_2 propellant**

  19 flow channels in each 3/4 in. HEX 52 in. long fuel element



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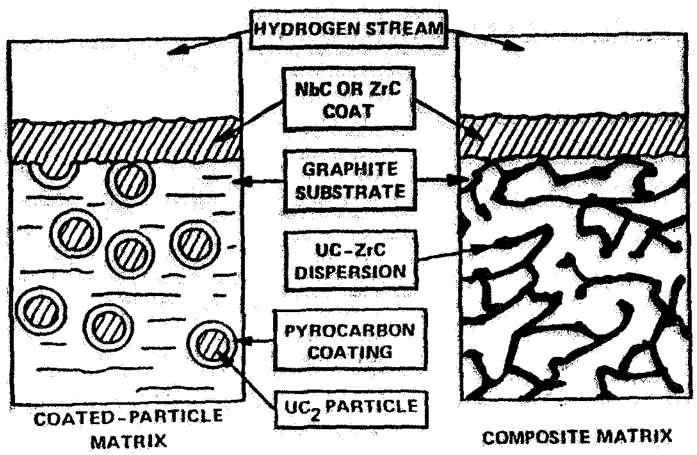
The NERVA/Rover prismatic graphite fuel element is 0.75 inch across the flats, and 52 inches long. It contains 19 flow holes (approximately 0.1 inch in diameter). All graphite surfaces have a protective ZrC or NbC layer to protect it from the hydrogen.

UC_2 fuel beads coated with pyrocarbon are dispersed through the matrix at a maximum fuel loading of 630 mg/cc. For the more recent composite type fuel element a maximum fuel loading of 700 mg/cc is achievable.

Nuclear design of the NERVA reactor limits the reactivity loss to approximately 1$ at the end of fuel life. Since the reactivity loss is mostly a result of loss of carbon due to the hydrogen corrosion, protective coatings are used to reduce the rate of carbon loss.

Fuel Element Comparison



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For all the NRX reactor and engine tests, the graphite-type fuel was used. However, toward the end of the NERVA/Rover program composite fuel emerged as the most promising candidate in reducing the hydrogen corrosion and in increasing the temperature capability of the prismatic fuel element.

The composite fuel element consisted for a dispersion of UC-ZrC web in the graphite substrate. Since this web is continuous, and essentially unaffected by hydrogen, it acts as a barrier and limits the carbon loss from the fuel.

Major Milestones in Fuel Development

- Graphite Fuel Element/HED NbC Coating (NRX-A2/A5)
- Graphite Fuel Element/HED NbC + Molybdenum Coatings (NRX-A6/XE')
- Graphite Fuel Element/GEM NbC/ZrC Coatings (PEWEE)
- High CTE Graphite Composite Fuel Element/GEM ZrC Coating (Nuclear Furnace -1)
- Carbide Fuel Element (Nuclear Furnace -1)

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The standard graphite fuel element with a HED NbC coating was used on NRX-2A/5A reactor series. The HED coating process resulted in a coating with a significant number of cracks, which seemed to have an adverse effect on the mid-band corrosion protection. In order to improve the mid-band corrosion performance of these elements, a molybdenum overcoat was applied to the fuel for NRX-A6/XE prime reactors.

The next improvement in the coating technology came with the lower temperature coating process, GEM, whereby ZrC or NbC coating could be applied without cracks in the coating. Fuel elements with this coating process were run in Pewee, but resulted in significant mid-band corrosion.

The fuel elements for the Nuclear Furnace-1 (NF-1) were of the high CTE graphite composite type with GEM ZrC coating, which were predicted to have eliminated the mid-band corrosion based on non-nuclear corrosion testing. Pure (U,Zr)C fuel elements were also tested in the nuclear furnace. These were manufactured as small hexagonal rods with a single cooling channel in the center. The carbide fuel elements were projected to have very low corrosion rates and very much higher temperature capability than both the graphite and the composite fuel elements.

NERVA/Rover Fuel Performance

REACTOR	FUEL EXIT TEMP. (K)	TIME AT MAX TEMP.(min)	AVG.WEIGHT LOSS PER ELEMENT (g)	TOTAL REACTIVITY LOSS (C)
NRX-A2	>2200	3.4	0.7	12
NRX-A3	>2400	16.5	16.5	58
NRX-EST	>2400	28.6	31.5	320
NRX-A5	>2400	30.1	27.1	223
NRX-A6	>2556	62.7	13.2	70
NRX-XE*	>2400	10.3	7.3	-
PEWEE-1	2750	43	20	
NF-1	2450	109	13.7	
PHOEBUS 1B	2445	30	13.7	

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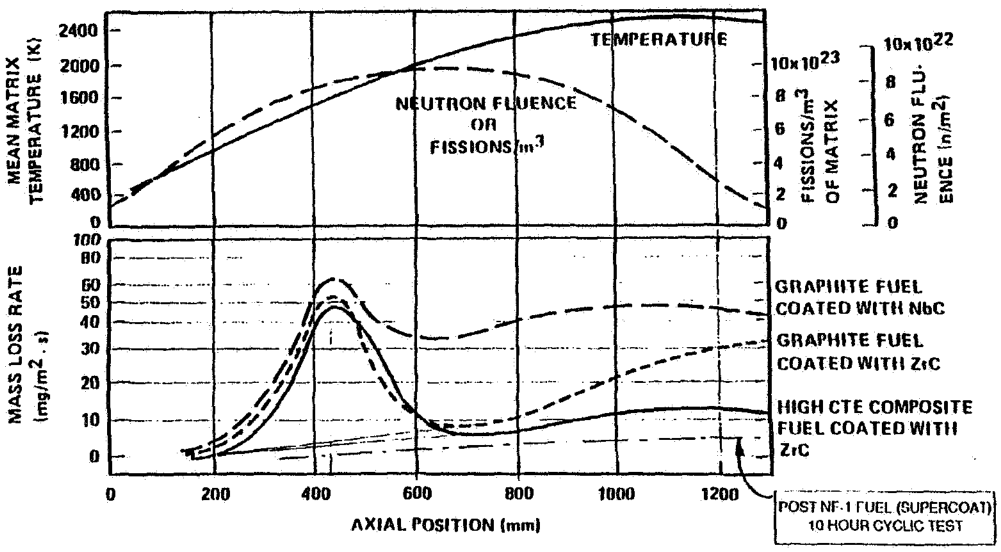
As a result of the improvements in the corrosion resistance of the fuel elements, the NERVA/Rover reactor tests showed a gradual increase in temperature capability and time at maximum temperature. The fuel life is dependent on the weight loss for the elements, and the resulting reactivity loss. Based on a reactivity margin of 1$ for corrosion from the fuel, the NERVA/Rover fuel life corresponds to a 15 to 20 g fuel element weight loss.

Corrosion of Rover/NERVA Fuel Elements



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The corrosion behavior along the length of the fuel element showed two different characteristics. From an axial position of 200mm to approximately 650mm from the cold end, an enhanced corrosion (called the mid-band corrosion) dominated. The temperature regime for this mechanism is 1000 to 2000 K, significantly below the maximum fuel temperature. In the progression of coating and fuel element improvements, there seemed to be negligible improvement in mid-band corrosion except for the demonstrated benefit of the molybdenum overcoat. From approximately 650mm to the hot end of the fuel element (called the hot end corrosion), the corrosion rate seemed to temperature related, and a significant decrease in the corrosion rate was observed as the coatings were improved. Electrically heated fuel element corrosion tests performed after the NF-1 testing demonstrated further improvements in the hot end corrosion rate, including a 10-hour life of a fuel element demonstrated by Westinghouse.

Key Reference Points For Fuel Experience

REACTOR TEST	TEMP FUEL EXIT (K)	TIME (MIN)	CYCLES	TOTAL LOSS (G)	MIDBAND (G)	HOT END (G)
NF-1 *	2444	108.8	4	13.7	8.6	5.1
NRX-A6 **	2556	62.7	1	12.8	2.3	10.5
NRX-XE' **	~2450	10.3	28	7.3	0.6	6.7

* Replacement composite fuel elements with crack free ZrC coating (GEM)
** Graphite fuel elements with NbC coating and molybdenum overcoat

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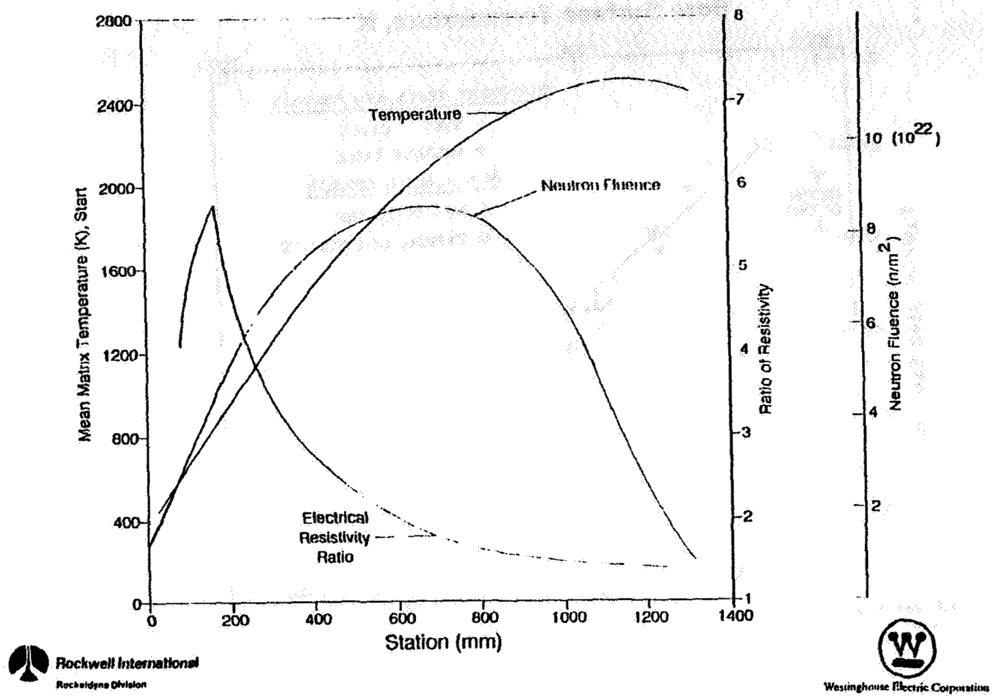
The most successful graphite fuel elements were those tested in NRX-A6, which were also used in NRX-XE prime engine configuration. These fuel elements utilized the HED NbC coating with molybdenum overcoat, and demonstrated a significant reduction in the mid-band corrosion compared to earlier NRX series tests.

The alternative fuel element technology is the composite, which was tested in NF-1. These elements, which were called the "replacement elements," were high CTE graphite coated with a superior ZrC coating (free of initial cracks) applied by GEM process.

The weight loss results for the A-6 and the XE prime fuels indicate that the A-6 vintage fuel has a significant sensitivity to thermal cycling. The NF-1 composite fuel elements demonstrated better hot end corrosion than the A-6 graphite fuel; however, a surprising degree of mid-band corrosion was still present.

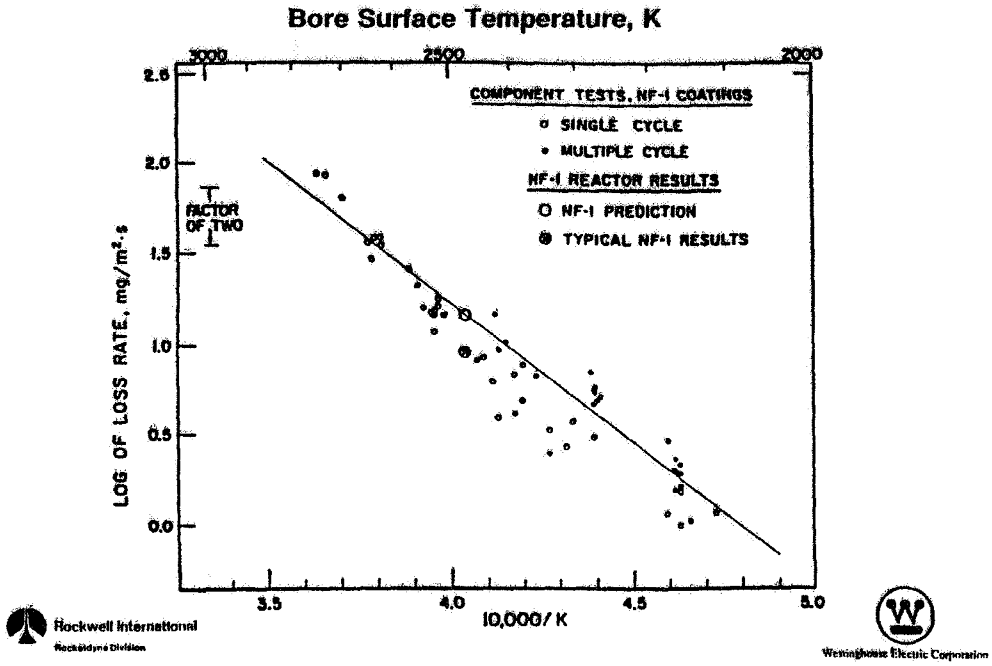
NF-1 Fuel Damage Explanation



Mid-band corrosion did not occur in the electrical testing of the composite fuel elements for NF-1, but caused the most significant weight loss during the reactor testing. Mid-band corrosion is believed to be a result of decreased thermal conductivity, possibly caused by fission fragment damage to the graphite matrix. The reduced thermal conductivity results in higher thermal gradients and increased thermal stresses, which causes cracking of the protective coatings and allows hydrogen to react with the graphite substrate. Mid-band corrosion must be fully understood and suppressed to meet performance requirements of the Fast Track program. Use of a molybdenum overcoat on composite fuel elements, or improved fuel particle coating in the graphite fuel to trap the fission fragments, are potential design solutions to mid-band corrosion.

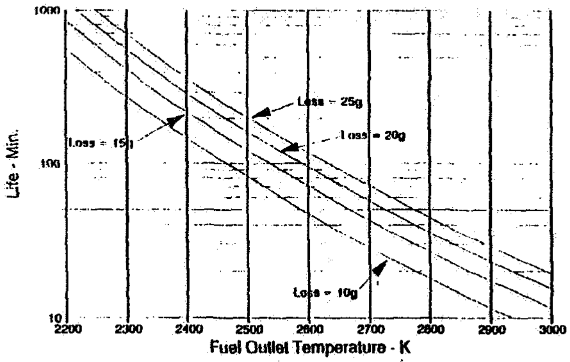
Composite Fuel Element

Hot End Corrosion Rate



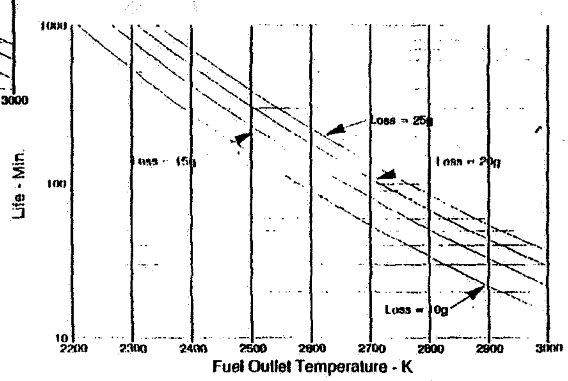
Composite fuel element testing provided a good correlation between the hot end corrosion measured in electrical testing and that observed in the NF-1. Hot end corrosion is caused by carbon diffusion through a protective coating and, therefore, is sensitive to the integrity of the coating, the coating thickness, and the temperature of the coating and fuel substrate.

Near Term Fuel Element



Graphite (A6 Technology)

Composite (NF1 Technology)



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Based on the assumption that the mid-band corrosion will be suppressed in near-term fuel elements, and the hot end corrosion rates measured in electrical testing and NF-1 testing, performance limits for near-term composite fuel can be calculated. Similar performance data can also be generated for the NRX-A6 type graphite fuel.

Comparing the projected near-term graphite fuel performance NRX-6A type with the composite fuel (NF-1 type) shows a 100-120 K temperature advantage for the composite fuel.

The improved performance of composite fuel is attributed to either the projected improvements in corrosion due to the composite fuel form or improved coatings used for NF-1 fuel elements. The improved coatings of NF-1 fuel elements are considered the most likely contributor to improved fuel performance.

Summary and Conclusions

- 1972 Fuel Technology was making progress toward meeting life/performance specifications consistent with current Fast Track requirements
  - understanding of midband corrosion was being developed
  - excellent hot end corrosion protection (ZrC on high CTE graphite) was demonstrated
- Corrosion limit for fuel elements was established based on 1$ reactivity loss
  - for NERVA type reactors, this translates into 15 to 20 grams corrosion loss per element
- Near term fuel development must resolve midband corrosion problem
  - fission fragment damage to graphite may be reduced by beaded fuel in graphite and composite matrix
  - molybdenum overcoat may suppress midband corrosion
  - improved graphite matrix may reduce or eliminate problem
- Near term composite fuel will have 4.5 hours life at 2470K to 2520K fuel outlet temperature
  - near term graphite fuel based on GEM ZrC/high CTE graphite is expected to perform similarly to near term composite fuel
- Near term fuel elements are expected to provide ISP ~850 seconds

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Assessment of Nuclear Safety Issues

- **Nuclear Safety Policy Working Group (NSPWG) Recommendations**
- **Accidental Criticality Sources for NERVA Derivative Reactor Design**
- **NERVA Safety Approach**
- **SP-100 Safety Approach**
- **NERVA Derivative Safety Approach**

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An assessment of the nuclear safety issues for a nuclear thermal propulsion system must be made based on the current regulatory guidelines, and the recommendation from the Nuclear Safety Policy Group (NSPWG). Starting with the accidental criticality sources for the NERVA derivative reactor design, the safety approached developed for the NERVA flight engine and the current SP-100 reactor safety approach, and the planned NERVA derivative safety approach will be discussed.

Assessment of Nuclear Safety Issues
From NSPWG & NP002 Safety Recommendations

- **No inadvertent reactor startup**
  - **Zero power testing on ground**
  - **Startup after achieving planned orbit**

- **No inadvertent criticality**
  - **Subcritical under all credible accident conditions**
  - **Highly reliable control system**

- **No significant radiological release or exposure**
  - **Only zero power testing prior to achieving planned orbit**
  - **29CFR1910.96 dose limits to flight crew**
  - **Insignificant impact to population of Earth**
  - **Insignificant impact on Earth and space environment**
  - **Spacecraft not rendered unusable when crew survives accident**
  - **Radiological release not impair use of spacecraft**

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Assessment of Nuclear Safety Issues
NSPWG Safety Recommendations (Cont'd)

- **No planned reentry**
  - **Minimize probability of inadvertent reentry**
  - **Minimize consequences of inadvertent reentry (high alt. disposal or intact reentry)**
  - **Subcritical at all times**
  - **Minimize impact dispersion**
- **Minimize hazardous materials release**
- **Ensure safe disposal**
  - **Part of mission planning**
  - **Adequate and reliable cooling, control and protection**
  - **Ensure non-premature final shutdown**
- **Safeguard nuclear material**
  - **Positive measures to prevent theft, diversion, loss or sabotage**
  - **Features to enhance safeguards and permit proven methods to be employed**
  - **Positive measures or features for recovery including jettison and tracking**

The NSPWG recommendations for safety requirements and guidelines addresses the protection of the public, the crew, the environment (both Earth and space environment), and includes recommendations for the safe disposal of the spent reactor system.

Assessment of Nuclear Safety Issues

Accidental Criticality Sources and Potential Countermeasures

Accidental Criticality Sources:

Source	Maximum Reactivity Insertion
Core Compaction	~ \$6 (80% Theoretical Density)
External Neutron Reflection	≈ \$3
Control Drum Roll-Out	\$3 (\$4.50 Drum Span)
Hydrogen Insertion	≈ \$83*
Water Immersion	≈ \$76

Potential Countermeasures:

C/M	Negative Reactivity Worth
Central + Peripheral Poison Wires	≈ \$90
Central Poison Wires Only	≈ \$10
Control Drums "Locked" Full-In	\$1.50 at Ambient Temperature
Safety rods	≈ \$90

*Limiting reactivity addition if the core could be completely flooded with high density LH_2.

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The sources for accidental criticality of a NERVA derivative reactor are core compaction, external neutron reflection (from water or soil), control drum rollout, hydrogen flooding, and water immersion. The countermeasures for reactivity events must assure a subcritical condition with a negative reactivity margin of 1\$.

For the NERVA reactors the criticality margin was assured using poison wires (7 for each fuel element). Other reactivity control means for NERVA-type reactors are the control drums or the possible introduction of safety rods within the core.

Assessment of Nuclear Safety Issues

NERVA Safety Approach

- Poison wires in core after assembly for shipping
  - ~7 Boron/aluminum wires/elements
  - Wires would be removed before launch
- Redundant safety features to preclude drum roll out
  - Permanent magnet stepping motor used in control drum drive actuator
  - Drum rollout requires erroneous command signal and closing electrical power circuit
- Anticriticality Destruct System (ACDS)
  - To fracture reactor by use of explosives
  - No more component greater than 3 fuel element
- Prevention of hydrogen insertion
  - Closing PFS valves when flooding is detected within 300 seconds of full leakage

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For the NERVA reactors the poison wires were primarily used to maintain the fully assembled and fueled reactor in a safe condition during transportation from the assembly area in Large, Pennsylvania, to NRTS. For a flight reactor the poison wires were to be removed prior to launch. Redundant safety features were used to preclude drum roll-out prior to the planned reactor startup in a safe orbit. To preclude criticality events for a launch accident or an inadvertent reentry event, an Anticriticality Destruct System (ACDS) would be used to break up the core.

Hydrogen flooding of the core was precluded using redundant valves and hydrogen sensors.

Assessment of Nuclear Safety Issues
SP-100 Safety Approach

- **Two redundant shutdown systems**
  - **Moveable reflector segments**
  - **Multiple safety rods**

    Only planned use for ultimate shutdown
- **Rhenium liner at core periphery to absorb thermal neutrons**
  - **Water immersion**
- **Inadvertent reentry and Earth impact**
  - **Reactor remains intact and subcritical**

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The SP-100 space power reactor system (SPRS) has been subjected to more extensive safety evaluations based on current guidelines. The decisions made and planning for the SP-100 SPRS will most probably apply to the NTR.

The SP-100 safety approach employs two redundant systems, moveable reflectors, and safety rods. The safety rods are designed to provide for permanent shutdown of the SP-100 reactor system after the completed mission. However, the safety rod design allows for the retraction of the rods from an unplanned insertion.

In addition to the moveable reflectors and safety rods, the SP-100 reactor includes a rhenium liner internal to the reactor vessel to capture neutrons thermalized external to the vessel and precludes back reflection from a water or earth immersion event. The SP-100 safety approach includes reactor system design features to assure an intact inadvertent reentry and earth impact event.

Assessment of Nuclear Safety Issues

Safety Approach for NERVA Derivative Reactor

- **Preliminary safety evaluations have been initiated**
- **Current safety guidelines appears to require dual shutdown systems**
  - **Control drums for normal operation**
  - **Safety rods for ultimate shutdown**
- **As part of the safety study, the design team is evaluating**
  - **Retractable safety rods**
  - **Neutron absorption at core periphery for Earth and water immersion**
  - **Impact of intact reentry**

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There has been no in-depth safety evaluation of the NERVA derivative reactor system completed to date. However, it is expected that the results of such an evaluation will be similar to SP-100 safety approach adapted to the reactor design. Based on the current safety guidelines, incorporation of dual or redundant safety shutdown systems will be needed to meet today's requirements.

As part of an ongoing safety evaluation for the NERVA derivative system Westinghouse will evaluate the use of retractable safety rods in the core and neutron absorbing liners at the core periphery to achieve the current safety guidelines. The design impact of an intact reentry will be evaluated for the reactor design.

Reactor Development Summary and Conclusions

- **Engineering and analysis of NERVA derivative reactors were successfully benchmarked against the NERVA/Rover test reactors for:**
  - **Reactor size and neutronics performance**
  - **Design characteristics such as fuel loading, ZrH moderator requirements, and control drum span**
  - **Internal shielding performance**
  - **Thermal performance of tie tubes**
- **A reactor conceptual design for the NASA 50K fast track engine was validated neutronically and thermally by analysis.**
- **The 1972 NERVA/Rover fuels technology must be recaptured and demonstrated.**
- **Near term fuel technology must resolve the mid-band corrosion problem.**
- **Near term fuel technology will meet fast track requirements.**
- **NERVA/Rover safety shutdown systems appears inadequate for today's requirements.**
  - **A secondary shutdown system will be developed for the NERVA derivative reactor designs.**

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In this project we performed trade-off studies, developed a single point conceptual reactor design, and validated this design thermally and neutronically.

The engineering and analysis supporting the trade-off studies and the point design were successfully benchmarked against the NERVA/Rover reactor designs. The reactor size and neutronic performance was established for a range of reactor sizes for engines producing 25 K to 75 K lbs thrust. The design characteristics such as fuel loading requirements and radial loading profile, ZrH moderator requirements, control drum worths and control span, were established.

A reactor concept for the 50 K lbf engine was developed and validated neutronically and thermally by analysis.

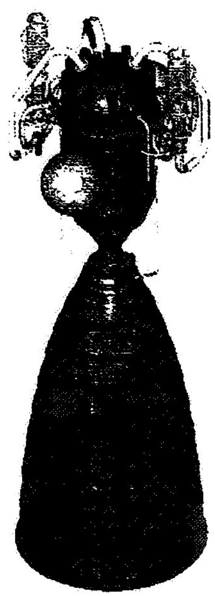
Internal shielding performance was established for the standard R-1 shield configuration, an unshielded, and for a reduced shield reactor.

The tie tube thermal performance was modelled, and evaluated for steady state conditions. Trade studies will establish the range of ZrC insulation properties and thermal transient performance.

The fuel technology of 1972 (the end of the NERVA program) was evaluated. This technology must be recovered and demonstrated as a baseline. Further, this technology must be advanced by eliminating or suppressing the midband corrosion problem to meet the fuel life requirements for the proposed missions. This "near term" fuel technology will meet the needs of the fast track program.

Reviewing the current requirements and recommendations for nuclear safety for the NTR, and the approach taken by other space power reactor systems, leads to the conclusion that the NERVA/Rover safety approach must be upgraded. Current plans are to evaluate a secondary shutdown system for the NERVA derivative reactor design.

50K PEWEE-DERIVED DUAL TURBOPUMP ENGINE



T_c 2550 K

SPLIT-FLOW EXPANDER CYCLE

ϵ 200:1

% L 110%

I_s 870 SEC

T/We 5.3 (W/SHIELD)

L_e 7.66 M

D_e 2.44 M

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50K PEWEE-DERIVED DUAL TURBOPUMP ENGINE

The 50Klb.-thrust engine is based on Rover/NERVA reactor core technology. Average fuel element exit gas temperature and core power density milestones were established by the Pewee reactor tested at NTS to a power level of 500 Mw in December 1968. The average fuel element exit gas temperature was approximately 2550K (4600R) with an average 1.18 Mw per fuel element. The Pewee reactor was also the first to incorporate zirconium hydride moderator in the tie-rod core support elements. Regeneratively-cooled tie-tubes rather than dump-cooled tie-rods were used in the Phoebus-2 reactor tested at NTS to a power level of 4,000 Mw in July 1968. Westinghouse used these important characteristics and results in establishing core preliminary design for the 50K reactor and the companion 25 and 75K reactors.

Dual turbopumps are used to provide an element of redundancy, as were used in the NERVA R-1 engine design. Valves provide isolation for a shutdown turbopump. The turbines are powered through a split-flow expander cycle where energy is derived from cooling the tie-tubes and nozzle in parallel.

Based on envelope considerations, the nozzle expansion ratio was set at 200:1 and the bell contour length at 110% of that for a 30° conical nozzle. For the 200:1 expansion ratio, this length provides the optimum nozzle thrust coefficient for hydrogen, resulting in a specific impulse of 870 seconds.

Hydrogen is bled from the turbine exhaust and regulated to provide engine pneumatic power and stage tank pressurization. Hydrogen is stored in two tanks to provide pneumatic power for engine restart. Pressurization of the stage Liquid Hydrogen tank is not required for starting due to 2-phase pumping capability of the pumps. If LH_2 tank pressure is below approximately 35 psia, then boost pumps are probably required.

The engine thrust-to-weight ratio is 5.3, including shielding which provides approximately half the maximum radiation field requirements of NPO. Without shielding, the T/We is 5.8.

Provisions for thrust vector control have been made by incorporation of a gimbal bearing at the top of the pressure vessel dome, and by two orthoganal outriggers for accepting gimbal actuators. The outriggers are mounted to the upper portion of the pressure vessel. Motion of the pump inlet during gimballing is accommodated by the scissor bellows, similar to those used on J-2 engines.

The engine length from the top of the gimbal bearing to the exit plane of the nozzle is 7.66M. The exit diameter of the nozzle is 2.44M.

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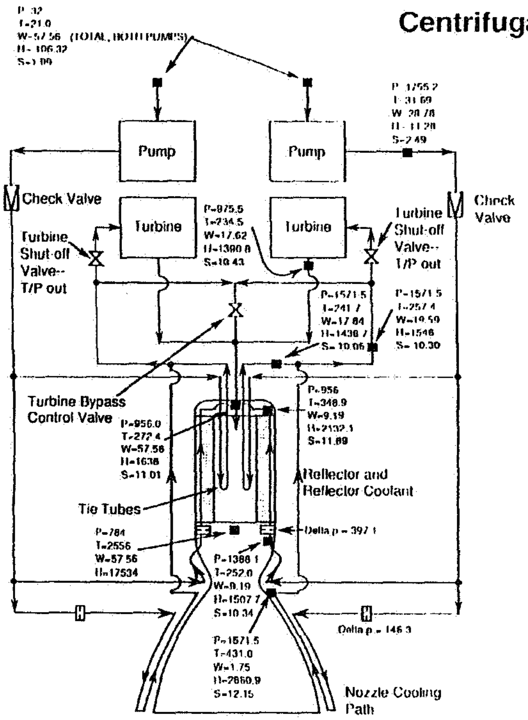
50K NTR, Expander Cycle, Dual T/P*
Centrifugal Pump



DESIGN VALUES:

PUMP FLOWRATE (TOTAL)	57.56 LB/SEC
PUMP DISCHARGE PRESSURE	1,755 PSIA
NUMBER OF PUMP STAGES	2
PUMP EFFICIENCY	72.58 %
TURBOPUMP RPM	47,500 RPM
TURBOPUMP POWER (EACH)	3,870 HP
TURBINE INLET TEMP	257.4 K
NUMBER OF TURBINE STAGES	1
TURBINE EFFICIENCY	72.56 %
TURBINE PRESSURE RATIO	1.611
TURBINE FLOW RATE (EACH)	17.62 LB/SEC
REACTOR/ENGINE THERMAL POWER	1,031.7 MW
FUEL ELEMENT TRANSFERRED POWER	965.1 MW
CORE THERMAL POWER (FUEL ELEMENT+TIE TUBE)	1,019.6 MW
ENGINE THRUST	50,000 LBF
NOZZLE CHAMBER TEMPERATURE	2,556 K
CHAMBER PRESSURE (NOZZLE STAGNATION)	784 PSIA
NOZZLE EXPANSION AREA RATIO	200:1
NOZZLE PERCENT LENGTH	110%
VACUUM SPECIFIC IMPULSE (DELIVERED)	868.72 SEC

Heat loads are as follows: Nozzle-con (total):	29.44 MW
Nozzle-div (total):	9.86 MW
Reflector (total):	12.10 MW
Tie-Tubes (total):	54.50 MW

P = PSIA
T = DEG K
W = LB/S
H = BTU/LB
S = BTU/LB-R

*Note: Flows indicated are for one-half of system.

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50K NTR, EXPANDER CYCLE, DUAL T/P CENTRIFUGAL PUMP

The 50K full-thrust system balance propellant conditions are shown at key points on the schematic. Significant design values and component heat loads are presented in the tables.

The pump inlet pressure was set at 32 psia, allowing a tank pressure of approximately 35 psia. The engine flowrate is 57.56 lb/sec (28.78 lb/sec for each pump). The pump pressure is 1,755 psia, resulting in a turbopump power of 3,870 Hp (each). Approximately 60% of pump flow goes to the tie-tubes and approximately 5% goes to the nozzle jacket, providing cooling and energy to power the turbine. The balance of the pump flow cools the chamber jacket and the reflector before joining the flow exhausted from the turbines.

The nozzle jacket (nozzle-div) and tie-tubes provide total power of approximately 64 Mw to heat the turbine drive loop to 257K with a turbine inlet pressure of 1571 psia. The single stage turbine has a flowrate of 17.6 lb/sec and a pressure ratio of 1.61 to drive the pump at 3,870 Hp and 47,500 rpm. Approximately 10% of turbine flow is bypassed around the turbine through a control valve to provide overall engine control in conjunction with the reactor control drum actuators.

The turbine exhausts and turbine bypass flows are combined and discharged into the pressure vessel dome where they join with the flow which cooled the chamber jacket and reflector. The total engine flow of 57.56 lb/sec then cools the fuel elements of the reactor with a power of 965 Mw. Hydrogen exits the reactor at 2556K (4600R) and 784 psia. This expands through the 200:1 expansion ratio, 110% length bell nozzle, providing a specific impulse of approximately 869 sec and thrust of 50,000 lb.

A check valve function is provided at each pump discharge and a shut-off valve function at turbine inlet so that a malfunctioning turbopump can be isolated while maintaining engine operation. These functions may be satisfied by a series/parallel arrangement of valves as was done with the NERVA R-1 engine. Likewise, the schematic indicates only a single turbine bypass control valve. The arrangement of valves to provide redundancy and meeting "Nuclear Thermal Rocket Engine Requirements," NASA N.P. #002, has not yet been addressed.

SPECIFIC IMPULSE ADVANCEMENT

•	XE-PRIME	710 SEC
	Tc 2270K BLEED CYCLE (TURBINE 10% FLOW) NOZZLE ε OF 10	
-	REGEN COOL FUEL ELEMENT SUPPORTS AND CORE PERIPHERY	+35
-	INCORPORATE EXPANDER CYCLE	+35
-	INCREASE ε TO 200, 110%L	+50
-	INCREASE Tc TO 2550K (PEWEE)	+40
•	PEWEE-BASED, GRAPHITE ELEMENT ENGINE I_s	870 SEC
-	INCREASE Tc TO 2700K	+30
•	COMPOSITE ELEMENT, ENGINE I_s	900 SEC

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SPECIFIC IMPULSE ADVANCEMENT

The XE-Prime is the baseline for nuclear thermal rocket engines, since it is the only engine configuration ever tested. This experimental engine was tested during much of the year in 1969, but full power and performance were most notably achieved in June when a chamber temperature of 2270K (4090R) was achieved. Due to the facts that a low expansion ratio (10:1) ground test nozzle was used, and that a bleed cycle was used to power the turbine which exhausted 10% of the engine flow at low specific impulse; an engine specific impulse of only 710 seconds was realized. Specific impulse is increased by 35 seconds by using regeneratively cooled (tie-tube) fuel element supports in place of dump-cooled, tie-rod fuel element supports, and by using regenerative cooling instead of dump-cooling in the core periphery where the transition is made from the irregular boundary of the hexagonal fuel elements to the circular boundary of the seal segments for sealing and bundling the core.

Specific impulse is increased by 35 seconds by using the expander cycle where the turbine exhaust is combined with the balance of the engine flow and the total flow is exhausted at the high reactor exit temperature rather than using the bleed cycle where the turbine flow (10% of the engine flow) is exhausted at low temperature and degrades engine specific impulse.

Specific impulse is increased by 50 seconds by increasing the nozzle expansion ratio from the experimental ground test engine value of 10:1 to 200:1 expansion ratio for a flight engine and using a 110% bell contour which provides the optimum thrust coefficient for hydrogen at this expansion ratio.

Specific impulse is increased by 40 seconds by increasing reactor exit gas temperature from the 2270K (4090R) of XE-Prime to the 2550K (4600R) of the Pewee Reactor test.

Cumming the above advancement results in the Pewee-Based, Graphite Element, Engine Specific Impulse of 870 seconds, since the Pewee Reactor used graphite fuel elements.

Specific impulse is increased by an additional 30 seconds if composite fuel elements where a reactor exit gas temperature of 2700K (4860R) can be achieved based on data from Nuclear Furnace, are used rather than the graphite fuel element with a reactor exit gas temperature of 2550K (4600R) based on data from Pewee Reactor testing. This results in the Composite Element (Nuclear-Furnace-Based) Engine Specific Impulse of 900 seconds.

ENGINE LENGTH AND NOZZLE SIZING

25K ENGINE

- STAGE REQUIREMENTS
  - ENGINE LENGTH - 6.0M
  - ENGINE I_s - 870 SEC
- CHAMBER PRESSURE INCREASED
  - FROM 621 PSIA (PEWEE)
  - TO 784 PSIA
  - MEETING REQUIREMENTS AND RESULTING IN:
    - ε 200:1
    - 110% LENGTH

50 AND 75K ENGINES

- USED 25K NOZZLE PARAMETERS
  - ε 200, 110%L

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ENGINE LENGTH AND NOZZLE SIZING

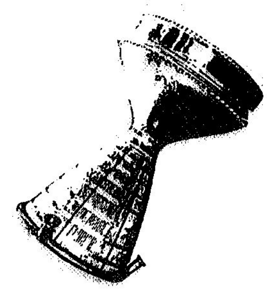
Initial effort in the program covering 25, 50 and 75K thrust engines was directed on the 25K engine, since stage requirements were provided for this engine. Engine length was limited to 6.0 meters with a specific impulse of 870 seconds.

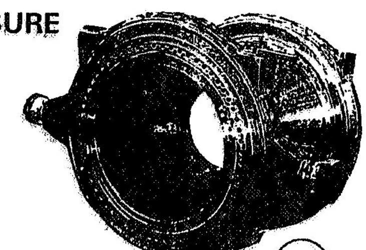
To meet the stage requirements, the chamber pressure of 621 psia from the Pewee test condition had to be increased to 784 psia. The higher pressure is beneficial to the reactor core with regard to heat transfer and pressure drop. The resulting nozzle has an expansion area ratio of 200:1 and a bell contour length of 110% of that for a 30° conical nozzle. With hydrogen, the 110% length provides maximum nozzle thrust coefficient for an area ratio of 200:1.

For the 50 and 75K engines, the same nozzle parameters of 200:1 expansion ratio and 110% length were used to result in a consistent family of engines from the standpoints of envelope, performance and weight.

NON-NUCLEAR COMPONENT TECHNOLOGY

- CHAMBER TECHNOLOGY
  - ROVER-KIWI, PHOEBUS
    - INCONEL-X TUBES
    - INCO 718 SHELL
    - FURNACE BRAZED ASSEMBLY
    - LIGHTWEIGHT
    - HIGH TEMP CAPABILITY
  - SSME
    - SLOTTED FORGED NARLOY
    - ELECTRODEPOSITED CU/NI CLOSURE
    - INCO 718 SHELL
    - HIGH HEAT FLUX CAPABILITY
    - LOW WALL TEMPERATURE
    - SUPERIOR LIFE CAPABILITY





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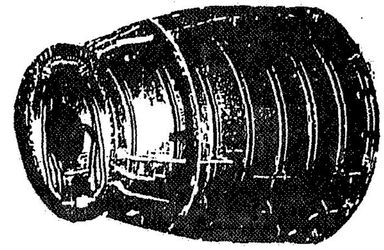
NON-NUCLEAR COMPONENT TECHNOLOGY
CHAMBER TECHNOLOGY

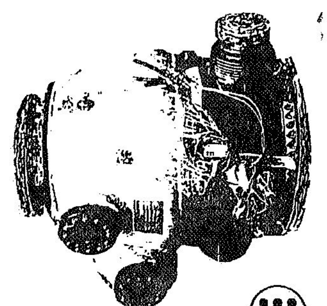
Rocketdyne has two technologies applicable to the NTR Chamber, the convergent and low area ratio divergent component which attaches to the bottom of the pressure vessel and ducts the reactor exit gas through the sonic region, delivering it to the high expansion ratio nozzle. One technology comes from earlier rocket engine programs, including the Rover program where tubular-wall chambers were employed, and still are today for engines such as Atlas and Delta. The other technology is the slotted, one-piece, copper-wall chamber used for the SSME.

The Rover tubular-wall chambers (as shown in the photo) were used for 7 of the 19 reactors tested in the Rover/NERVA program, including Phoebus 1B at conditions approaching 1500 Mw, 750 psi chamber pressure, and throat heat flux of 30 BTU/in^2 sec. These chambers have a contraction ratio of approximately 20 to interface with the reactor at an inlet diameter of approximately 35 inches, and an expansion ratio of 12 to exhaust into the atmosphere at NTS conditions. Inconel-X tubing was used with an Inconel 718 one-piece forged Shell/Flange. The chamber was a furnace-brazed assembly. This technology provides a lightweight chamber with approximately 1000K (1760R) wall temperature capability.

The SSME slotted, one-piece, copper-wall chamber (as shown in the photo) was developed for and used on all Space Shuttle Main Engines. Three SSME's on each Space Shuttle flight have now powered over 50 missions, and flight configuration engine testing exceeds 120 hours. The SSME chamber operates at a chamber pressure of approximately 3000 psi with a wall temperature at the throat of approximately 800K (1460R) and heat flux of approximately 100 BTU/in^2 sec. The chamber has a contraction ratio of approximately 3 and an expansion ratio of 5 with a throat diameter of approximately 10 inches. The slotted, forged NARloy (Rocketdyne copper alloy) chamber liner is electrodeposited on the outer envelope with a thin copper and then heavier nickel closure of the coolant slots. A welded Inconel 718 shell, manifold and flange assembly complete the chamber. This technology provides high heat flux (100 BTU/in^2 sec) capability with low (1000F) wall temperature. Although the weight is somewhat higher for an NTR chamber than with the tubular-wall Rover chamber technology, the SSME chamber technology is favored due to superior Life-Cycle capability and general robustness.



- NOZZLE TECHNOLOGY
  - SSME
    - A-286 TUBES
    - FURNACE BRAZED ASSEMBLY
- TURBOPUMP TECHNOLOGY
  - INDUCER
    - Mk 15F, Mk 25
      - 2-PHASE PUMPING CAPABILITY
      - TITANIUM
  - IMPELLERS
    - SSME HPFTP
    - TITANIUM
  - BEARINGS - HYDROSTATIC
    - Mk25, Mk29FD
  - TURBINE
    - TITANIUM, A-286, OR 718



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NON-NUCLEAR COMPONENT TECHNOLOGY (CONT'D.) NOZZLE AND TURBOPUMP TECHNOLOGY

Rocketdyne high-expansion-ratio, regeneratively-cooled, nozzle technology is exemplified by the SSME nozzle shown in the photo. The construction is tubular-wall, using A-286 tubes, furnace-brazed assembly, in order to reduce the weight of this large nozzle. The nozzle inlet is an area ratio of 5 with an exit area ratio of 77.5. The nozzle length is approximately 10 ft. with an exit diameter of approximately 7-1/2 ft. The nozzle employs approximately 1,000 thin wall, A-286 tubes. As with the chamber, this SSME technology provides capability beyond the requirements of the NTR, resulting in a robust design.

Rocketdyne technology applicable to the NTR turbopump draws on elements from several programs; however, is best exemplified by the Mark 29F (shown in the photo) which was developed as the liquid hydrogen turbopump for the J-2S engine. Rocketdyne initiated design and development of large, liquid-hydrogen turbopumps in 1958 under the Rover program for application to nuclear rockets. Successful testing of the first large liquid-hydrogen (Mark 9) pump in 1960 allowed commitment to the J-2 engine which used the Mark 15F (derived from Mark 9) axial, liquid-hydrogen pump. The Mark 9 and evolutionary Mark 25 turbopumps were used for 11 of the 19 reactors tested in the Rover/NERVA program and in the PlumBrook B1 NTR cold-flow engine simulation teststand.

Inducer technology for liquid-hydrogen pumps is exemplified by 2-phase testing of the Mark 15F and Mark 25 at inlet vapor volume fractions of up to 30%. Low flow-coefficient, larger diameter inducers were then fabricated for these pumps and tested to even higher vapor volume fractions. This capability provides for pumping of liquid hydrogen from a saturated tank without the need for pressurization to provide net positive suction head at the pump inlet. This provides weight savings to the stage in tankweight, pressurant and storage tankweights, and vented propellant weight.

The liquid hydrogen centrifugal pump technology of the Mark 29F was advanced with the SSME High Pressure Fuel Turbopump. Significant improvement in efficiency was achieved.
Hydrostatic bearings were demonstrated in the Mark 25 pump in testing in 1972 at NTS. These bearings used interior rolling element bearings which provided the rotation a lower speeds during the slow start-up and very slow shut-down associated with NTR's. This arrangement considerably reduces the DN requirement and life requirement for the rolling element bearing and allows use of radiation-resistant cage materials. Pure hydrostatic bearings in liquid hydrogen is an ongoing development with the Mark 29FD.

Due to the low inlet temperature (approximately 300K) and single stage of the expander cycle turbine, the turbine technology for the NTR is simplified compared to the high temperature, multi-state turbines developed for most rocket engines. Areas of concern are hydrogen embrittlement and hydriding in which Rocketdyne has much of the world's applicable experience.

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ORIGINAL PAGE IS
OF POOR QUALITY

50K PEWEE-DERIVED DUAL-TURBOPUMP NTR ENGINE CHANGING REACTOR SUPPORT RATIO SIGNIFICANTLY IMPROVES T/W_e

	3:1 FUEL ELEMENT/ SUPPORT RATIO	Δ WEIGHT 3 TO 6:1 RATIO	6:1 FUEL ELEMENT/ SUPPORT RATIO
REACTOR	8,200	-1,950	6,250
NOZZLE	1,200	+20	1,220
TURBOPUMP	270	+10	280
LINES AND CONTROLS	860	+60	920
SHIELD	1,100	-250	850
	11,630 LB	-2,110 LB	9,520 LB
ENGINE T/W	4.3		5.3

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50K PEWEE-DERIVED DUAL-TURBOPUMP NTR ENGINE CHANGING REACTOR SUPPORT RATIO SIGNIFICANTLY IMPROVES T/We

Between the conceptual sizing of the 50K reactor and preliminary sizing, Westinghouse determined through nuclear analysis that a 6:1 Fuel Element to Support Element Ratio could be used rather than a 3:1 ratio, resulting in higher power density and less weight for the 50K reactor. Thus, the 50K core is more similar to the 75K core, which uses a 6:1 support ratio (as used in KIWI-B4, NRX and Phoebus reactors), rather than the 25K core, which uses a 3:1 support ratio (as used in Pewee).

Elimination of virtually half the supports (with their Zirconium Hydride moderator, tie-tubes and graphite parts), together with the core and reflector diameter reduction effects, results in a reactor weight reduction of approximately 2,000 lb. or approximately 25%. The shield likewise decreases approximately 25% due to the reduction in diameter.

The non-nuclear components increase slightly (approximately 5%) in weight due to increase in pump discharge pressure to provide higher pressure ratio to drive the turbine as a result of lower turbine inlet temperature because of reducing the number, and therefore, total power of the tie-tubes (one contained in each support element) by 50%.

Due to the reduction in engine weight as the result of basically cutting the number of support elements in half (going from 3 to 6:1 Fuel Element to Support Element ratio), the engine weight is reduced by approximately 20%, and therefore, the engine thrust-to-weight ratio improves by 20%.

LIFE IMPACT ON CHAMBER TEMPERATURE AND SPECIFIC IMPULSE ADVANCEMENT

• XE-PRIME		710 SEC	
Tc 2270K BLEED CYCLE (TURBINE 10% FLOW) NOZZLE ϵ OF 10			
- REGEN COOL FUEL ELEMENT SUPPORTS AND CORE PERIPHERY		+ 35 SEC	
- INCORPORATE EXPANDER CYCLE		+ 35 SEC	
- INCREASE ϵ TO 200, 110% L		+ 50 SEC	
- INCREASE Tc WITH GRAPHITE FUEL ELEMENT TO:		2550 K	2450 K
	Life	1.5 HR	4.5 HR
	ΔI_s	+ 40 SEC	+ 20 SEC
• GRAPHITE ELEMENT ENGINE I_s		870 SEC	850 SEC
- INCREASE Tc WITH COMPOSITE ELEMENT TO:		2700 K	2550 K
	Life	1.5 HR	4.5 HR
	ΔI_s	+ 30 SEC	+ 20 SEC
• COMPOSITE ELEMENT, ENGINE I_s		900 SEC	870 SEC

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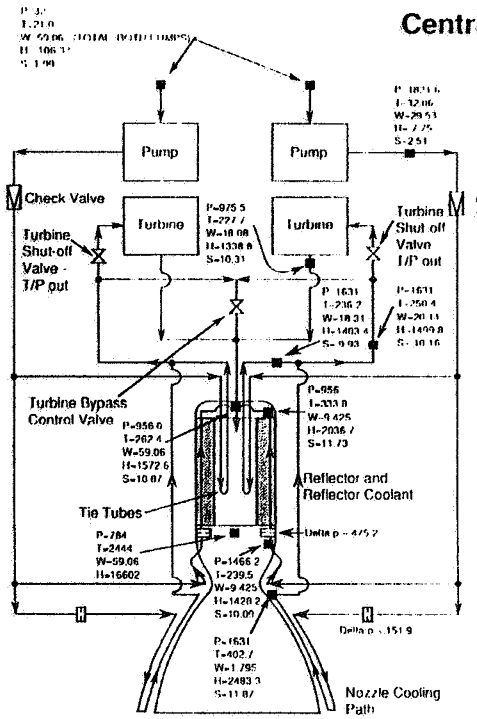
LIFE IMPACT ON CHAMBER TEMPERATURE AND SPECIFIC IMPULSE ADVANCEMENT

As in the prior chart, "Specific Impulse Advancement," the XE-Prime is the baseline for specific impulse at 710 sec. Also, as in the prior chart, advancements by 1) regenerative cooling of core structure (+35 sec), 2) using the expander cycle (+35 sec), and 3) using the 200:1 expansion ratio nozzle (+50 sec), increase specific impulse by 120 seconds.

However, Westinghouse evaluation of Rover/NERVA Fuel Element Mass Loss resulted in life capability of 1.5 hours for the prior chart's Graphite Fuel Element at Pewee Average Exit Gas Temperature of 2550K and resulting specific impulse of 870 sec, and Composite Fuel Element Gas Temperature of 2700K with specific impulse of 900 sec. This 1.5 hour data is presented in the left hand column. "Near-Term" engine life requirements are for a Life Capability of 4.5 hours. To meet the 4.5 hour Life with the allocated reactivity loss of 1%, translates into 18 to 20 grams mass loss per element and the resulting temperatures and specific impulses as shown in the right hand column. For the 4.5 hour life requirement, the resulting Graphite Element Engine Specific Impulse is 850 sec, and the Composite Element Engine Specific Impulse is 870 seconds.

50K NTR, Expander Cycle, Dual T/P*
Centrifugal Pump



DESIGN VALUES

PUMP FLOWRATE (TOTAL)	59.06 LB/SEC
PUMP DISCHARGE PRESSURE	1,022 PSIA
NUMBER OF PUMP STAGES	2
PUMP EFFICIENCY	72.58 %
TURBOPUMP RPM	47,500 RPM
TURBOPUMP POWER (EACH)	4,118 HP
TURBINE INLET TEMP	250.4 K
NUMBER OF TURBINE STAGES	1
TURBINE EFFICIENCY	71.71 %
TURBINE PRESSURE RATIO	1.672
TURBINE FLOW RATE (EACH)	18.08 LB/SEC
REACTOR/ENGINE THERMAL POWER	1,002.8 MW
FUEL ELEMENT TRANSFERRED POWER	936.2 MW
CORE THERMAL POWER (FUEL ELEMENT + TIE TUBE)	990.7 MW
ENGINE THRUST	50,000 LB
NOZZLE CHAMBER TEMPERATURE	2,444 K
CHAMBER PRESSURE (NOZZLE STAGNATION)	784 PSIA
NOZZLE EXPANSION AREA RATIO	200:1
NOZZLE PERCENT LENGTH	110%
VACUUM SPECIFIC IMPULSE (DELIVERED)	846.64 SEC

Heat loads are as follows:	**Nozzle-con (total):**	**28.55 MW**
	Nozzle-div (total):	**9.44 MW**
	Reflector (total):	**12.10 MW**
	Tie-Tubes (total):	**54.50 MW**

P = PSIA
T = DEG K
W = LB/S
H = BTU/LB
S = BTU/LB-R

***Note: Flows indicated are for one-half of system.**

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50K NTR, EXPANDER CYCLE, DUAL T/P CENTRIFUGAL PUMP

In conjunction with the reduction in graphite fuel element average exit gas temperature from a nominal 2550K to 2450K to meet the 4.5 hour Life requirement, a revised system balance was performed

Compared to the balance shown on the prior chart, the average reactor exit gas temperature (nozzle chamber temperature) is reduced by approximately 4% to 2,444K. This results in an approximate 2% reduction in specific impulse to 846.64 sec. To maintain engine thrust at 50,000 lbf requires increasing flowrate by approximately 2% to 59.06 lb/sec. The increase in flowrate and reduction in temperature result in an approximate 3% reduction in Reactor/Engine Thermal Power to 1,002.8 Megawatts.

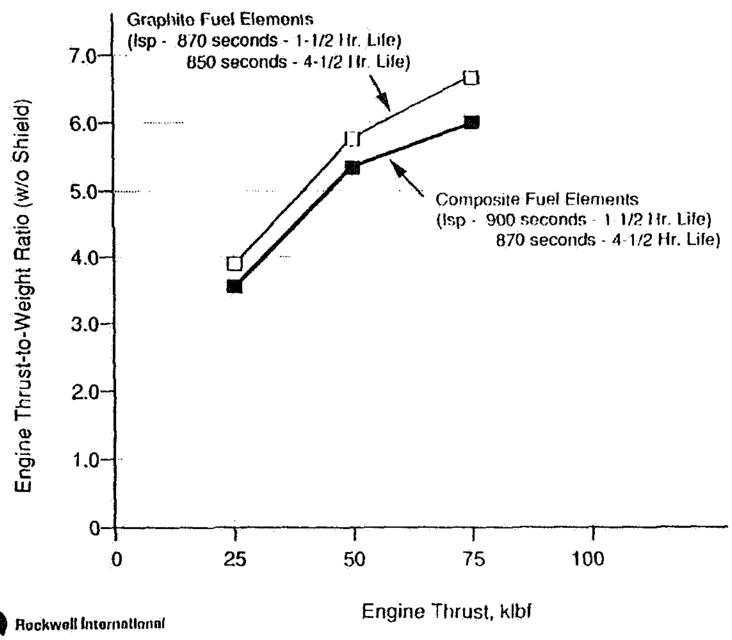
The reactor configuration for the 1.5 hr and 4.5 hr life would be the same. Fuel element thermal conditions and stresses actually reduce due to the 4% reduction in temperature and 3% reduction in power. Fuel element mechanical stresses stay the same since the reactor exit pressure is fixed (784 psia) and the core pressure drop is essentially the same due to the 2% reduction in velocity (2% increase in flowrate and 4% increase in density due to lower temperature) and 4% increase in density. So the reactor weight remains essentially the same between the 1.5 hr and 4.5 hr life cases.

The chamber and nozzle sizes remain the same, due to the 2% increase in flowrate and 4% reduction in temperature resulting in the same throat area.

The pump flowrate increases by 2% and the discharge pressure increases by 4% due to the 6% increase in turbine pressure ratio required to provide the 6% higher turbopump power. This results in a 4% increase in turbopump weight which is equivalent to approximately 0.1% in engine weight. Due to the 2% increase in flowrate and the 4% increase in pump discharge pressure, the turbopump line weight increases by 6% which is equivalent to approximately 0.4% in engine weight.

So the engine weight effect in going from the 1.5 to the 4.5 hour life is an approximate 0.5% increase in weight due to the 2% increase in flow and 4% increase in pump discharge pressure with the majority of the effect being due to the pump discharge and turbine lines.

Effect of Thrust and Fuel Element on Engine Thrust-to-Weight Ratio



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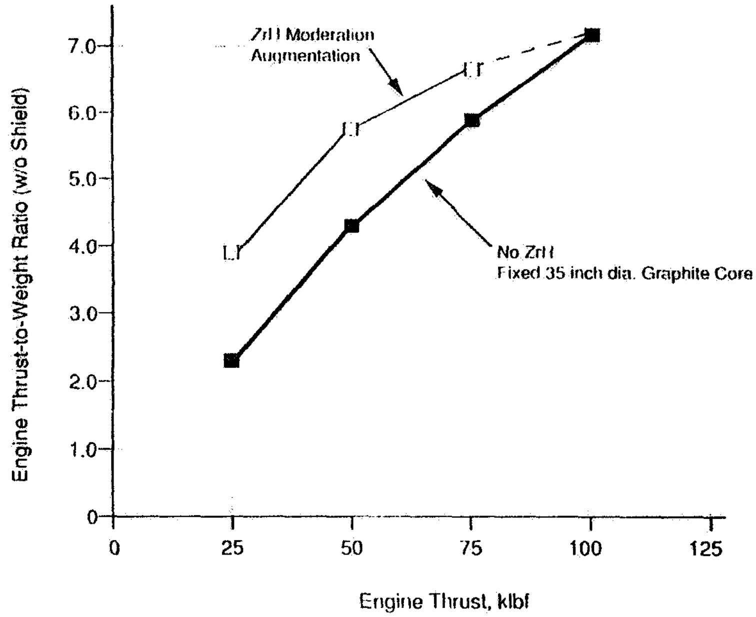
EFFECT OF THRUST AND FUEL ELEMENT ON ENGINE THRUST-TO-WEIGHT RATIO

The effect of thrust for the 25, 50 and 75K lb engines on engine thrust-to-weight ratio (without including radiation shielding) is shown for both Graphite and Composite Fuel elements.

As a result of discussion related to the previous chart regarding engine weight changes in going from 1.5 to 4.5 hr Life, the engine weight increases by approximately 0.5% primarily in line weight due to the 2% increase in flowrate and the 4% increase in pump discharge pressure. This is a negligible effect to these thrust-to-weight ratio plots. Therefore, the plot for each fuel element applies for the range of Life and Specific Impulse shown.

Effect of Thrust and Use of ZrH Moderator on Engine Thrust-to-Weight Ratio



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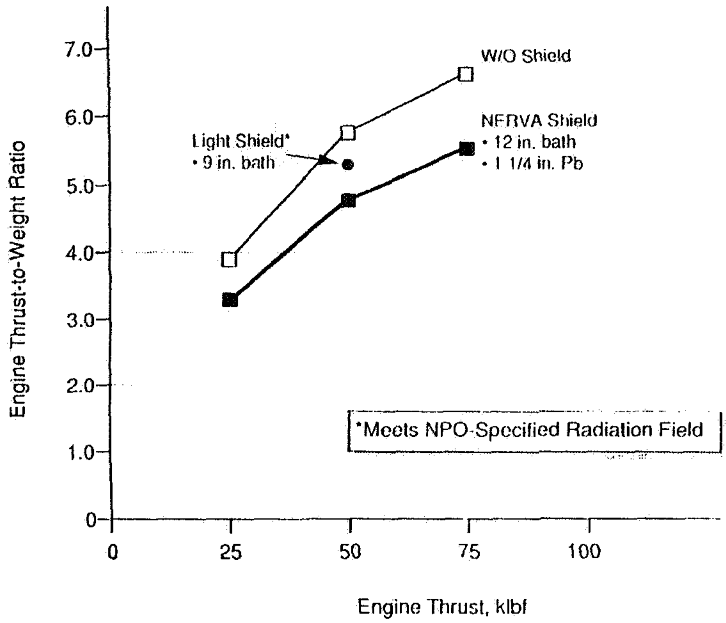
EFFECT OF THRUST AND USE OF ZrH MODERATOR ON ENGINE THRUST-TO-WEIGHT RATIO

The effect of Zirconium Hydride moderation on engine thrust-to-weight ratio (without including radiation shielding) is shown over the thrust range of 25 to 100Klb. The lower curve represents engines using a fixed 35-inch diameter, 52-inch long core containing no ZrH. The upper curve represents engines using reactors containing ZrH as necessary to minimize size and weight for the 25, 50 and 75Klb thrust reactors as analyzed by Westinghouse. Other Westinghouse preliminary analysis indicates that ZrH does not reduce the weight of a reactor with a 35-inch diameter core for a 100Klb thrust engine. On this basis, the dashed line was constructed between the 75Klb ZrH moderated point and the 100Klb point without ZrH.

So ZrH moderation provides no advantage at 100Klb thrust, approximately 10% weight advantage at 75K, approximately 35% weight advantage at 50K, and approximately 75% weight advantage at 25K thrust.

Effect of Shield on Engine Thrust-to-Weight Ratio



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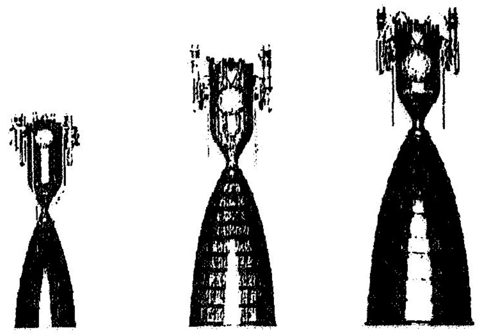
EFFECT OF SHIELD ON ENGINE THRUST-TO-WEIGHT RATIO

The effect of shield weight on engine thrust-to-weight ratio is shown. The upper data represents the engine thrust-to-weight ratio without including radiation shielding. The lower data represents engine thrust-to-weight ratio using the internal shield used for the NERVA R-1 engines; namely, 12 inches of BATH (Boron, Aluminum, and Titanium Hydride) and 1-1/4 inches of lead.

During the program, NPO specified neutron flux levels and gamma dose level to be met by the shielding for the "Near-Term" reactor. Due to concern about lead melting during decay heat removal, the lead was removed. The NPO-specified radiation field also allowed reduction in the BATH thickness from 12 inches down to 9 inches. The Westinghouse analysis of the resulting radiation field for the 50K engine results in neutron fluxes and gamma dose approximately half that specified by NPO, indicating that a small further reduction in BATH thickness may be made.

At the 50K thrust level, the Light Shield provides approximately 10% improvement in engine thrust-to-weight ratio over the NERVA shield. The light shield represents an approximate 9% reduction from the thrust-to-weight ratio of 5.8 for the unshielded 50K engine.

PEWEE - DERIVED NTR's



THRUST	25K	50K	75K
T/We (W/SHIELD)	3.6	5.3	6.0
Le	6.00M	7.66M	8.74M
De	1.73M	2.44M	2.99M

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PEWEE-DERIVED NTR'S

Conceptual designs were performed for 25, 50 and 75K thrust engines based on Rover/NERVA reactor technology. Fuel element power was based on Pewee test results of approximately 1.2 Megawatts per fuel element average. The engine thrust-to-weight ratio requirement of ≥ 4 with shielding was not met by the 25K engine which has a value of approximately 3.6 with a shield meeting the NPO radiation field requirements. *Unshielded, the 25K engine has a thrust-to-weight ratio of* approximately 3.9, so the requirement of 4 is not met even without the radiation shield.

Certainly, based on engine thrust-to-weight ratio, the preferred engine thrust would be 100K or more based on the elimination of the weight penalty associated with the use of Zirconium Hydride moderator and achieving a shielded engine thrust-to-weight ratio of approximately 6.5, or approximately 7.1 without radiation shielding.

NPO selected the 50K engine for more detailed analysis and specified radiation field values to determine the shield. The radiation field requirements allowed lightening the shield by approximately 900 lb resulting in an improvement in shielded *engine thrust-to-weight ratio from 4.8 to 5.3, with an unshielded thrust-to-weight ratio of 5.8.*

The 50K engine has an overall length of 7.66 meters and a nozzle exit diameter of 2.44 meters.

These parameters all apply regardless of whether 1) the engine life is 1.5 hours operating at a chamber temperature of 2550K with 870 sec specific impulse, or 2) the engine life is 4.5 hours operating at a chamber temperature of 2450K with 850 sec specific impulse.

Key Technology/Streamline Development Assessment

- Approach
  - Identify critical design areas and technology issues
  - Assess actions and program impacts
  - Determine critical path
- Areas addressed
  - Safety, hydrogen pumping, nozzle, valves, instrumentation and controls, reactor, engine system and test facility

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Key Technology/Streamline Development Assessment

In performing this assessment, design areas of the engine system were reviewed and critical technology issues were identified, together with actions required to address these issued and their impact on the program. A critical path was inferred from this analysis. The design areas addressed were safety, hydrogen pumping, the nozzle, valves, instrumentation and controls, the reactor assembly, and the engine system and test facility. In most instances it was found that recovering or referencing existing technology provides the design basis. However, several system design issues exist where new design solutions and test verifications would be required, and effort to resolve these items should be emphasized early in the program.

KEY TECHNOLOGIES ASSESSMENT

AREA OF DESIGN	TECHNOLOGY ISSUE	ACTION REQUIRED	PROGRAM IMPACT
Safety	Water immersion criticality	Analysis of several design options for reactivity control in thermal range; recover NERVA plans	Engine test facility to test as-designed engine, no additional cost or schedule
Safety	Intact reentry or total dispersal	Design for reentry heating; recover reentry data; consider engine/vehicle interactions	Verification testing

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Key Technologies Assessment

Ready technology from many sources forms the foundation for the Rocketdyne-Westinghouse NERVA-derived engine concept. In reviewing key technology areas the goal was to assess both the needed actions to resolve the particular issues and the programmatic impact. In many cases technology recovery was the principal action, and there was no programmatic impact. In a few areas, issues not anchored in ready technology were identifed, and their resolution should be addressed early in the program. We expect no intractable problems.

Safety issues in all phases of the program have to be adequately identified, and procedures and design solutions have to be qualified. Four safety issues are noted here: (1) water immersion criticality, (2) intact reentry or total dispersal, (3) the concern over flammability and dispersion of nuclear materials in a launch explosion and fire, and (4) the impact of the continued nuclear power generation of a shutdown engine in a cluster. The latter affects engine-cluster specific performance, but is also a safety issue because the overheating potentially can damage the stage placing the crew at risk, and ejection of the engine with its potential for generating debris may pose a threat to the stage or to future missions.

Restartability requires adequate systems for decay heat removal that do not consume excessive quantities of hydrogen. A flight-qualified decay heat removal system was never demonstrated in the Rover/NERVA program.

KEY TECHNOLOGIES ASSESSMENT

AREA OF DESIGN	TECHNOLOGY ISSUE	ACTION REQUIRED	PROGRAM IMPACT
Safety	Launch fire resistance of nuclear and hazardous materials	Analysis and design of fire retarding or resisting features such as plug in throat	Mockup test in simulated launch explosion/fire
Safety	Engine-out continued power generation; dead weight	Analysis of alternatives: auxiliary cooling, shielding, ejection, etc.	No additional impact
Restartability	Decay heat removal	Analysis and design of optimum method for conserving propellant	Test decay heat removal system during engine tests

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KEY TECHNOLOGIES ASSESSMENT

AREA OF DESIGN	TECHNOLOGY ISSUE	ACTION REQUIRED	PROGRAM IMPACT
Hydrogen pumping	Two-phase pumping	Design pumping system compatible with tank pressurization limits; recover Rover test data, Mark 25 and Mark 15 data	Test candidate configuration in pump test facility
Hydrogen pumping	Bearings	Design for 10 restarts, and slow start and shutdown transients; recover Mark 25 data with hybrid hydrostatic bearings, SSME experience, Mark 29FD with hydrostatic bearings	Demonstrate during pump qualification test
Hydrogen pumping	Seals	Select radiation-hard seal materials	Part of turbopump design and test

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Key Technologies Assessment

The hydrogen turbopumps, while based on mostly proven hardware, must be qualified for the radiation environment, 10 restarts in space, slow startup and shutdown transients, and 4.5 hours of accumulated full-power operation. The solutions to these issues are anchored in existing technology, but a rigorous test program will be required to demonstrate, adeqately, the design integrity.

Chamber and nozzle experience with the SSME satisfies most design requirements, except those dealing with the radiation environment, such as joint seal design. Testing of seals in a radiation environment would be required.

Radiation resistance of valves--bodies, stem, guides, actuators, seals, seats--must be incorporated in the design and verified by test, and turbine bypass valve functional performance must be assured by test.

KEY TECHNOLOGIES ASSESSMENT

AREA OF DESIGN	TECHNOLOGY ISSUE	ACTION REQUIRED	PROGRAM IMPACT
Nozzle	Radiation resistant joint seal	Test seal configuration in radiation environment	Need to identify test facility
Nozzle	High heat flux	Use SSME NARloy-Z slotted channel approach	No additional impact
Valves	Radiation resistance	Select radiation resistant materials; Recover data from Rover/NERVA, SP-100, LMFBR	Life-cycle test valves separately, and evaluate after engine test

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KEY TECHNOLOGIES ASSESSMENT

AREA OF DESIGN	TECHNOLOGY ISSUE	ACTION REQUIRED	PROGRAM IMPACT
Valves	Turbine bypass control	Modify SSME and J-2 valves and perform life-cycle test	Test in hydrogen flow facility
Reactor assembly	Fuel element midband corrosion	Develop and test coating materials and processes	Identify reactor test facility; test and evaluate in engine testing
Reactor assembly	Vessel design for intact reentry or dispersal, decay heat removal	Select compatible materials and configuration	Safety requirements drive the design

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Key Technologies Assessment

In the reactor assembly, midband corrosion found in the Rover/NERVA fuel elements was being addressed when the program was canceled. Resolution of this issue will be of prime importance at the outset, with in-pile testing of fuel elements and clusters necessary to validate the solution. The reactor vessel must be designed to meet the safety requirement of intact reentry or total dispersal in the event of an inadvertant reentry from space. The control drum actuators may feature electrical or pneumatic drives, or both, based on Rover/NERVA or current design technology. The support plate must contain the tie-tube inlet and outlet flow passages, operate with minimal thermal distortion, and be structurally robust. Data from the Phoebus 2A reactor and from the NERVA design would be the bases for the new design. To achieve higher operating temperatures and performance development of composite fuel would be continued from the Rover program baseline. The instrumentation and control design area would initially address key sensors and the engine health monitoring system. Current technology would serve these areas.

Finally, the ground testing of the complete engine system is the key step in qualification for piloted operation. An operational facility will be needed with adequate engine-exhaust scrubbing to meet environmental and safety concerns, and with well-designed altitude simulation diffusers and ejectors. Because design, the environmental approval process, construction, and acceptance testing will require about 6 years to complete, embarking on this effort almost immediately is essential to meeting the desired 10-year development goal. We believe that this facility is the critical path.

KEY TECHNOLOGIES ASSESSMENT

AREA OF DESIGN	TECHNOLOGY ISSUE	ACTION REQUIRED	PROGRAM IMPACT
Reactor assembly	Drum actuators	Incorporate Rover/NERVA or SP-100 designs; evaluate fluidic stepping motors	Testing required
Reactor assembly	Support plate with tie tubes	Evaluate Phoebus-2A and NERVA-R1 designs, fabricate and test unit	Test in hydrogen flow facility--parallel with pump testing
Reactor assembly	Higher temperature fuel	Develop composite fuel	Test reactor or nuclear furnace required
Instrumentation and controls	Hydrogen flow measurement	Evaluate candidate flow meters, including fluidics; procure candidates and test	Test in hydrogen flow facility

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KEY TECHNOLOGIES ASSESSMENT

AREA OF DESIGN	TECHNOLOGY ISSUE	ACTION REQUIRED	PROGRAM IMPACT
Instrumentation and controls	Reactor temperature, pressure	Incorporate Rover/NERVA and advanced reactor sensors	Life test in a reactor in hydrogen; evaluate in engine test
Instrumentation and controls	Health monitoring system	Incorporate Rocketdyne state-of-the-art diagnostics	Evaluate in engine system test
Engine system test	Scrubbing engine exhaust, diffuser and ejector technology; environmental concerns	Proceed with site selection and facility design and construction; recover NF-1 scrubber data, state of the art Rocketdyne diffuser/ejector technology	Critical path to engine qualification

Technology Assessment Results

- Technology available for most issues

  Rover/NERVA, SSME, Rocketdyne state of the art, SP-100, terrestrial advanced reactors, state-of-the-art electronics and computers

- Unresolved system design issues

  Loss of turbopumps, lifetime, intact reentry--water subcriticality (or total dispersal), decay heat removal, engine-out cooling during operations, fuel midband corrosion

- Critical path is engine test facility

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Technology Assessment Results

The assessment of key technologies led to the conclusions that (1) existing technology in reactors and engine systems is applicable to most design areas, (2) there are issues requiring attention early in the program to assure satisfactory resolution, and (3) the assured early availability of an engine/reactor test facility is critical to meet, successfully meet the 10-year engine qualification goal.

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Streamline Development Requires Early Agreement on Requirements

NASA prepares:	Rocketdyne/Westinghouse:
• Mission and performance requirements	• Provide comments on draft requirements and specifications
• Safety requirements	• Perform QFD analysis
• Interface control structure	• Prepare design criteria and test plans
• Engine specification	• Specify test facility needs

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Streamline Development Requires Early Agreement on Requirements

To establish a good foundation for a successful development program both NASA and the Rocketdyne/Westinghluse team must understand and accept the design requirements, program and technical interface requirements, and design criteria and testing needs. Poorly understood or shifting requirements can lead to delays and cost escalation. We believe that a QFD analysis of the program will lead to well-understood requirements and optimum design and hardware results.

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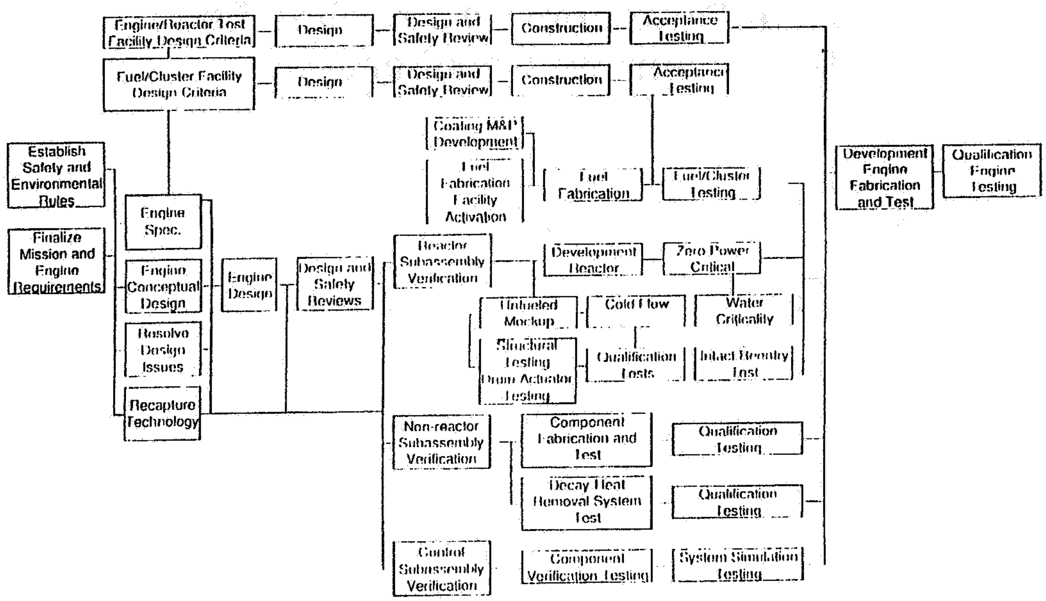
NTR Streamline Development Logic



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NTR Streamline Development Logic

A development logic diagram can include many layers of detail and be organized in many different ways. This high-level diagram shows many necessary tasks in setting requirements, recapturing technology, resolution of key design issues, facility design, construction, and activation, and testing of components and systems. The most important message is that the program must start with well-defined requirements and design criteria, and that the availability of key test facilities will drive the rate of achievement of the 10-year goals. Near-term activities of conceptual design, technology recovery, and resolution of design issues will provide a sound basis for proceeding quickly as substantial funding becomes available.

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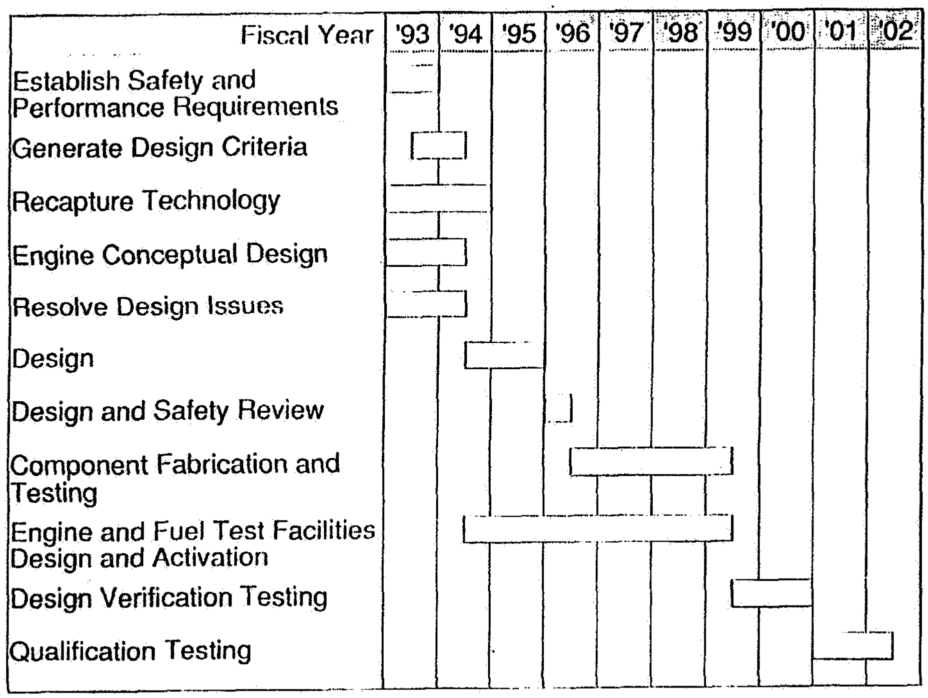
NTR Streamline Development Plan Summary



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10/15/92

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NTR Streamline Development Plan Summary

The time-phasing of key groups of activities from the development logic diagram shows that several tasks should be emphasized at the start: setting requirements, technology recapture, and establishing design criteria. Test facility design, construction and activation must also begin promptly to assure that the 10 year schedule can be met.

Advanced Propulsion Engine Assessment based on a Cermet Reactor

for the

Nuclear Propulsion TIM
October 20-23, 1992

Pratt & Whitney
Randy C. Parsley

25807

PRATT & WHITNEY DESIGN CHOICE BASED ON FUNDAMENTAL PRIORITIES

Priority		
1	Safety	Retention of fission products Robust design/simple operation Emergency operation
2	Reliability	Design life = 4X operational Fundamental material compatibility Positive coolant flow management Retention of fuel/stoichiometry Low developement risk
3	Cost	Ground qualification Exploration architecture
4	Performance	High thrust–to–weight High specific impulse

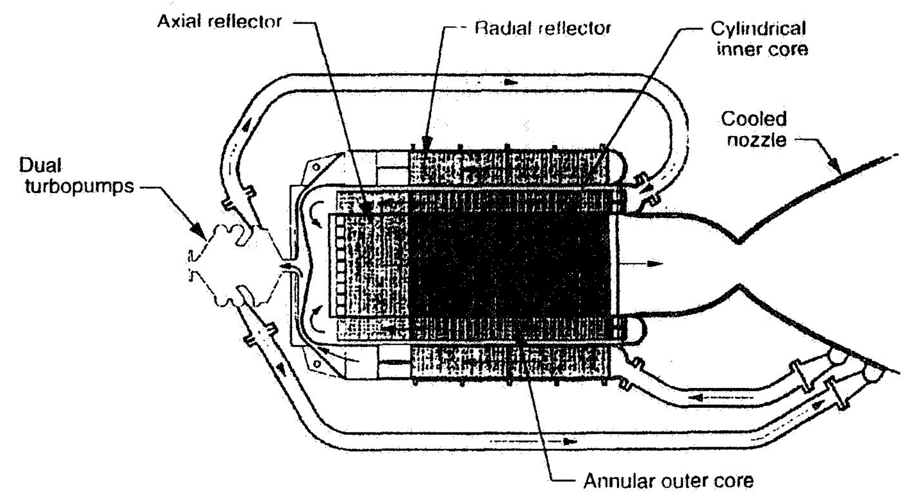
PRATT & WHITNEY XNR2000 CERMET NTRE

25609

<u>Pratt & Whitney Design Choice Based on Fundamental Priorities</u>

A preferred Pratt & Whitney conceptual Nuclear Thermal Rocket Engine, NTRE, has been designed based on the fundamental NASA priorities of safety, reliability, cost, and performance. The basic philosophy underlying the design of the XNR2000 is the utilization of the most reliable form of ultrahigh temperature nuclear fuel and development of a core configuration which is optimized for uniform power distribution, operational flexibility, power maneuverability, weight, and robustness. The P&W NTRE system employs a fast spectrum, cermet fueled reactor configured in an expander cycle to ensure maximum operational safety. The cermet fuel form provides retention of fuel and fission products as well as high strength for a simplified structural design and tolerance to power and temperature cycling. System reliability has been addressed by the use of cermet based fuels, moderate reactor temperatures, and a two-pass reactor flowpath. The cermet, refractory metal fuels provide fundamental material compatibility in the expected operating environment as well as retention of fuel and stoichiometry. The two-pass reactor has been designed to a 4X life requirement and provides positive coolant flow management. A baseline 25,000 lb thrust level is used to minimize ground qualification costs and maximize exploration mission applicability. Finally, the P&W NTRE has been designed to provide high specific impulse at a high thrust-to-weight level.

XNR2000 SYSTEM CONFIGURED AS AN EXPANDER CYCLE



PRATT & WHITNEY XNR2000 CERMET NTRE

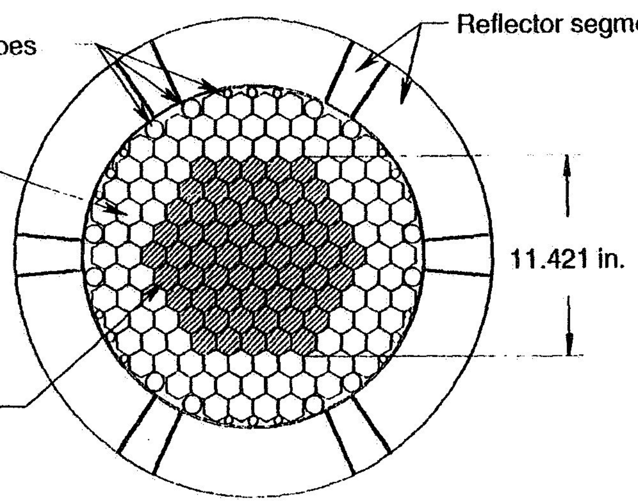
25411

XNR2000 System Configured As An Expander Cycle

The expander cycle was selected for the proven reliability, robustness, and high efficiency to meet NASA requirements. The XNR2000 expander cycle rocket engine uses heat pick-up in the nozzle, chamber, reflector regions, and regenerative cooling of the pressure vessel and upper plenum structure to drive the pumping system and deliver hydrogen to the lower plenum of the reactor core. The reactor heats the hydrogen in a two-pass flow configuration and delivers the hydrogen propellant to the nozzle chamber before expansion through a 200 area ratio nozzle. The reactor is comprised of an outer annulus core of 90 Mo-UO2 prismatic fuel elements and an inner cylindrical core of 61 W-UO2 prismatic fuel elements. An upper reflector, integral with the fuel elements, is used to provide axial neutron reflection and is comprised of BeO. An outer annulus of Be surrounds the reactor and serves as the radial reflector.

The baseline XNR2000 operates at propellant chamber temperature at 2670 K and chamber pressure of 766 psi to deliver 25,000 lbs. thrust at a specific impulse of 900 seconds.

REACTOR CONFIGURED FROM PRISMATIC FUEL ELEMENTS



PRATT & WHITNEY XNR2000 CERMET NTRE

25376

XNR2000 Reactor Is Configured From Prismatic Fuel Elements

A medial plane radial cross-section of the XNR2000 NTRE is shown. Looking down at the engine, hydrogen enters the outer annular ring of fuel elements (unshaded) flows up and is then directed through the inner cylinder of fuel elements (shaded) and flows down. The cross section displays the baseline control approach selected for the XNR2000. One possible option for providing redundant reactor shutdown control would be the insertion of Re rods inside the radial support tubes shown in the medial plane cross-section. The rods could be included in the design to prevent inadvertent reactor excursions during transportation, pre-launch, or booster transfer. The Re rods would provide an independent back-up safety mechanism but would not be used for reactor control.

CERMET FUELS WERE PURSUED FOR BOTH PROPULSION AND POWER

Early cermet failures are most remembered not later material successes

Program refocus to power applications reinforce low temperature bias

Successful demonstrations
- 10,000 hrs at 1,950k (in reactor)
- 1,000 hrs at 2,278k
- 3 hrs at 3,178k
- Transients to 2,879k at 10,000k/sec (in reactor)
- 37 hole element fabrication

PRATT & WHITNEY XNR2000 CERMET NTRE

25012

<u>Cermet Fuels were Pursued For Both Propulsion and Power</u>

The basic design philosophy used in the development of the XNR2000 was to employ the most reliable form of ultrahigh temperature nuclear fuel. The approach used to accomplish this goal was to make use of the extensive data and lessons learned in the ROVER/NERVA Nuclear Fuel and Reactor System Development Program, Argonne National Laboratory Nuclear Rocket Program, and the General Electric Advanced Nuclear Propulsion Project 710 Program. A summary of results of cermet fuel development programs of 1950's and 60's is published in two sets of reports: ANL-7236, (1968) "Nuclear Rocket Program", Terminal Report, GEMP-600, (1973), "7.0 High Temperature Gas Reactor Program Summary Report", Vols. I-VI.

P&W INTERNAL STUDIES IDENTIFIED CERMET APPROACH AS SUPERIOR

		Chemical stability		
Constituents	T_{melt}	Matrix	Clad	Hydrogen
UO_2	3100k	Solved*	Total	Total
Tungsten	3650k		Total	Total
Tungsten – Re	3400k			Total

Element

Strength – High
Conductivity – High
Ductility (Cold) – Adequate
(Hot) – Good

Clad/matrix CET match – Good
Matrix/fuel CET match – Good

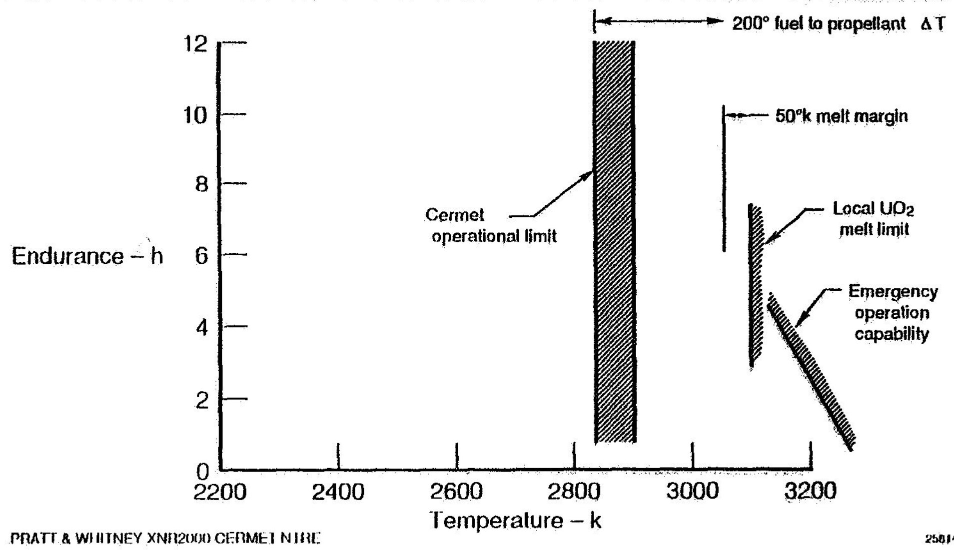
*UO_2 stabilized with 6% Gd or Th

PRATT & WHITNEY XNR2000 CERMET NTRE 25811

<u>P&W Internal Studies Identified Cermet Approach As Superior</u>

- Cermet fuel made of UO2 dispersed in Tungsten or Molybdenum cladded with Mo or W based alloys were tested at high temperature in both nuclear and non-nuclear environments and displayed superior performance in the expected operating environment of an NTRE. Retention of fission products and fuels, thermal shock resistance, hydrogen compatibility, high thermal conductivity, clad/matrix CET compatibility, and high strength are several major advantages of the cermet fuel form.

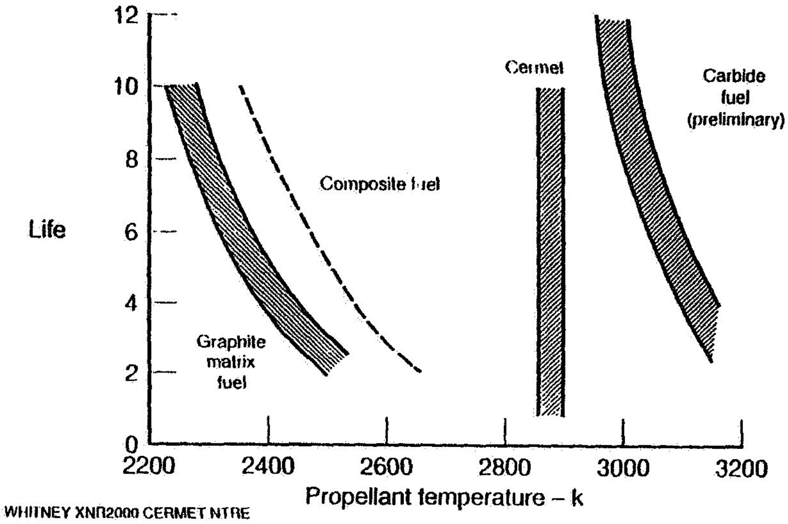
CERMET OPERATING LIMITS CAN BE ESTABLISHED FROM EXISTING DATA



<u>Cermet Operating Limits Can Be Established From Existing Data</u>

A critical review of the cermet fuel development programs was used to establish operating limits for the P&W XNR2000 reactor. The XNR2000 has a temperature margin of 250K using the local UO_2 melt temperature as a conservative upper limit on reactor temperature. The XNR2000 has an endurance on the order of 100's of hours at the selected operating temperature.

CERMET FUEL SHOULD ALWAYS BE INCLUDED ON THIS CURVE



PRATT & WHITNEY XNR2000 CERMET NTRE

25815

Cermet Fuel Should Always Be Included On This Curve

The predicted endurance of carbon based and cermet based fuels is shown as a function of propellant exit temperature. As shown in the Figure, the endurance of cermet fuels is independent of operating temperature up to the melt temperature of the fuel. However, the endurance of carbon based fuels is a function of propellant temperature because of stoichiometry changes due to chemical diffusion of carbon based fuels in a hot hydrogen environment. A change in the mechanical, thermal, and neutronic characteristics at Carbon based fuels decreases the fuel endurance with increasing operating temperature. The cermet fuels display constant characteristics because there is no fuel/matrix diffusion and the material stoichiometry is constant with temperature.

RESULTS OF NERVA/ROVER RESEARCH REACTOR TESTS WITH TEMPERATURE OVER 2222K

	Time at temperatures over 2222k (sec)	Maximum chamber temperature (k)	Time at max temperature (sec)	Reactivity loss (grams/element)
PHOEBUS–1A, EPIV (22 June 1966)	651	2367	5	?
PHOEBUS–1B (23 Feb 1967)	400	2292	5	13.7
PEWEE–1 (Dec 1968)		2555 (fuel exit temp)	2400	20
NF–1 (June 1972)		2444 (fuel exit temp)	6528	13.7

PRATT & WHITNEY XNR2000 CERMET NTRF

25859

Results of NERVA/ROVER Research Reactor Tests With Temperature Over 2222

A short summary of often quoted NERVA/ROVER test results. It should be noted that while time at temperatures over 2222 is high, and most often quoted, the time at maximum temperatures is often quite low. Additionally, fuel temperatures are often quoted rather than the lower hydrogen temperatures, adding to the confusion. Reactivity loss was proven to be a major concern in the NERVA/ROVER Program and could significantly increase the cost or even prohibit ground qualification.

RESULTS OF NERVA/ROVER DEVELOPMENT REACTOR TESTS WITH TEMPERATURE OVER 2222K

	Time at temp over 2222k (sec)	Max temp (k)	Time at max temp (sec)	Reactivity loss (cents)
NRX–A3 (23 April 1965)	5	2244	3	22.3
NRX/EST, EPIII (2 March 1966)	75	2292	5	2.5
NRX/EST, EPIV (16 March 1966)	110	2264	5	46.7
NRX/EST, EPIVA (25 March 1966)	816	2264	450	282.4
NRX–A5, EPIII (8 June 1966)	473	2286	7	22.5
NRX–A5, EPIV (23 June 1966)	873	2333	7	212.3
NRX–A6, EPIIIA (15 Dec 1967)	3764	2405	10	70
XE–PRIME, EP5 (March 1969)	10	2278	5	–

PRATT & WHITNEY XNR2000 CERMET NTRF 25800

PRATT & WHITNEY XNR2000 DESIGN TEAM

Project director: Randy Parsley (P&W)

Pratt & Whitney

Steve Peery (P.I. systems)
Russ Joyner (Missions)
Alan Dixon (Mech Des)
Samim Anghaie (Nuclear, T.H.)
Gerald Feller (Nuclear)
Mike Malone (Materials)
Paul Harris (Materials consultant)

Babcock & Wilcox

Kurt Westerman
Steve Scoles (P.I.)
Russ Jensen (Materials)
James Rhodes (Nuclear)

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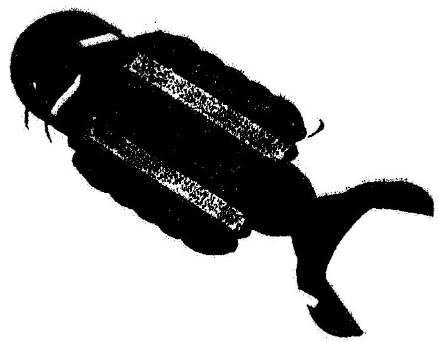
XNR2000 CONCEPT CONFIGURED TO ADDRESS P&W PRIORITIES



- Fast spectrum CERMET
- Dual pass reactor
- Robust and safe

PRATT & WHITNEY XNR2000 CERMET NTRE

25816

<u>XNR2000 Concept Configured To Address P&W Priorities</u>

A conceptual nuclear thermal rocket (NTR), the XNR2000, has been developed for manned space exploration missions. The discriminating features of the XNR2000 that provide attractive attributes are the use of Cermet fuel, a dual-pass reactor flowpath, and a simple robust cycle. An XNR2000 system description, reactor thermal hydraulic summary, and throttling, operating temperature, and thrust size effects will be presented. This package presents the summary of a 6 month NASA funded study to develop and assess concept feasibility, thrust level range implications, and manned mission impacts of an NTR system based on a prismatic Cermet reactor.

XNR2000 CONFIGURED TO MEET NASA REQUIREMENTS

I_{sp} > 850 sec (at 200 area ratio)

T/W > 4

Throttling 25% at rated temperature

Single burn duration 60 min (max)

Engine life > 270 min at rated thrust

Remain subcritical upon impact and immersion

ALARA fission product release

Dual turbopump arrangement

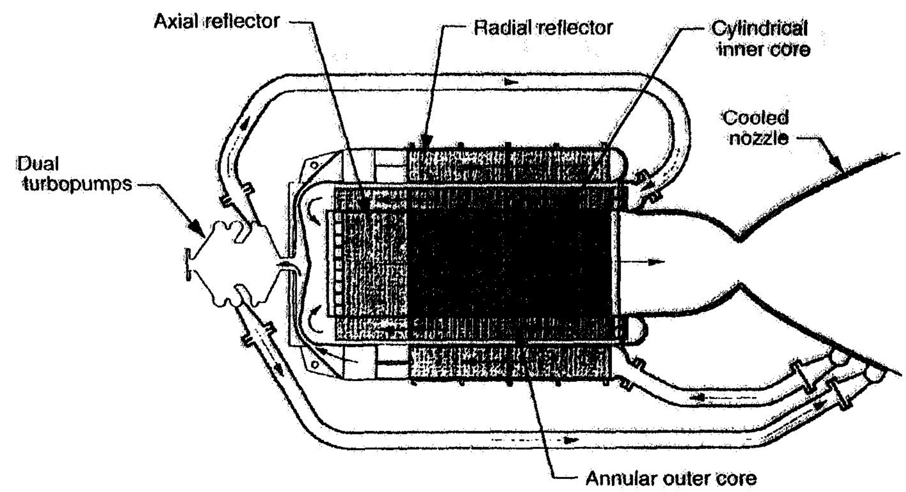
25k, 50k & 75k Thrust size

PRATT & WHITNEY XNR2000 CERMET NTRE 25374

XNR2000 Configured To Meet NASA Requirements

The XNR2000 Nuclear Thermal Rocket Engine, NTRE, was configured to meet or exceed the performance requirements of a man-rated NTR System. The propulsion requirements listed are described in detail in the "Nuclear Thermal Rocket Engine Requirements" document, version 3 February 10, 1992. The baseline thrust size was set at 25,000 lb. and thrust size effects were determined for engines of 50,000 and 75,000 lb. of thrust. Safety and reliability are key NTRE propulsion requirements for the manned-mission Space exploration applications and were considered foremost in the conceptual design of the XNR2000. The reactor fuel and spectrum selection was specifically dictated by the ALARA fission product release and reactor subcritically requirements.

XNR2000 SYSTEM CONFIGURED AS AN EXPANDER CYCLE



PRATT & WHITNEY XNR2000 CERMET NTRE

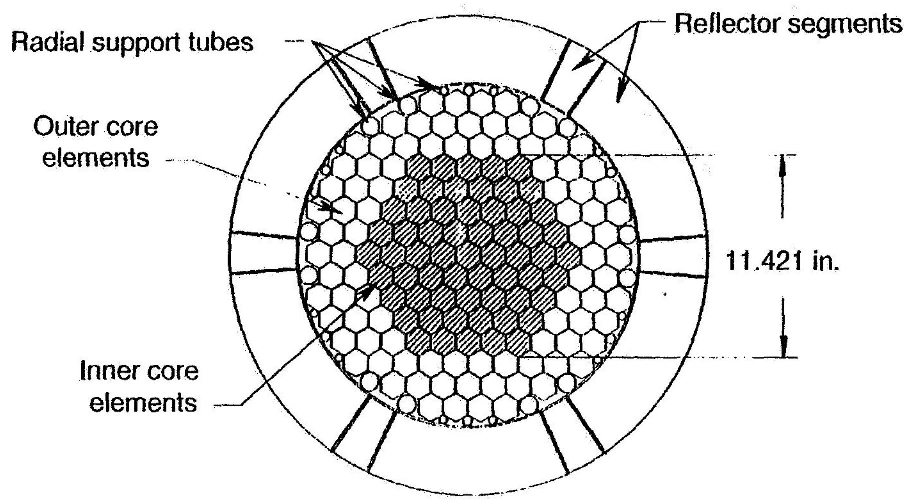
25411

<u>XNR2000 System Configured As An Expander Cycle</u>

The expander cycle was selected for the proven reliability, robustness, and high efficiency to meet NASA requirements. The XNR2000 expander cycle rocket engine uses heat pick up in the nozzle, chamber, reflector regions, and regenerative cooling of the pressure vessel and upper plenum structure to drive the pumping system and deliver hydrogen to the lower plenum of the reactor core. The reactor heats the hydrogen in a two-pass flow configuration and delivers the hydrogen propellant to the nozzle chamber before expansion through a 200 area ratio nozzle. The reactor is comprised of an outer annulus core of 90 Mo-UO2 prismatic fuel elements and an inner cylindrical core of 61 W-UO2 prismatic fuel elements. An upper reflector, integral with the fuel elements, is used to provide axial neutron reflection and is comprised of BeO. An outer annulus of Be surrounds the reactor and serves as the radial reflector.

The baseline XNR2000 operates at propellant chamber temperature at 2670 K and chamber pressure of 766 psi to deliver 25,000 lbs. thrust at a specific impulse of 900 seconds.

REACTOR CONFIGURED FROM PRISMATIC FUEL ELEMENTS



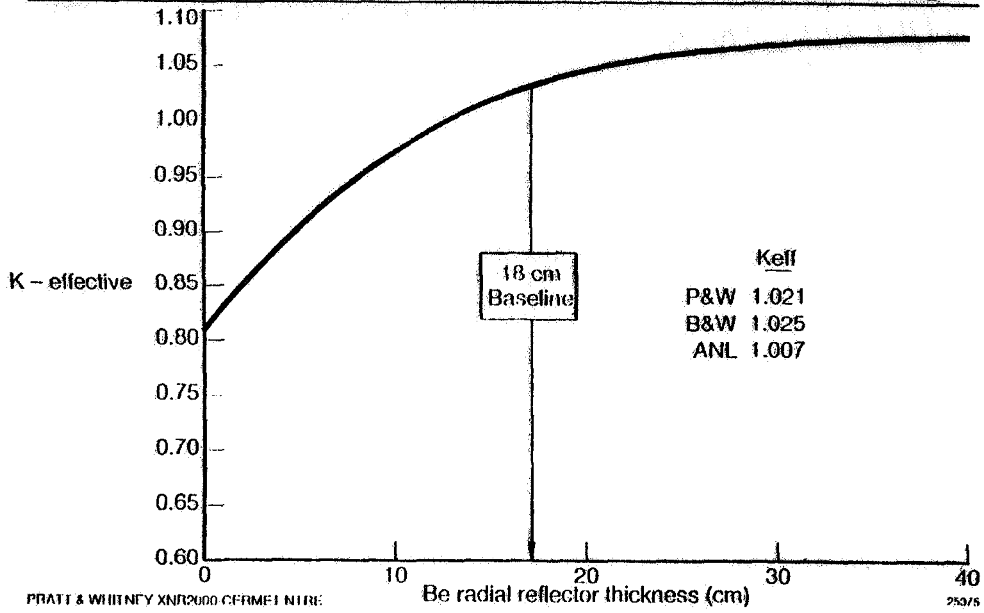
PRATT & WHITNEY XNR2000 CERMET NTRE

25378

<u>XNR2000 Reactor is Configured From Prismatic Fuel Elements</u>

A medial plane radial cross-section of the XNR2000 NTRE is shown. Looking down at the engine, hydrogen enters the outer annular ring of fuel elements (unshaded) flows up and is then directed through the inner cylinder of fuel elements (shaded) and flows down. The cross section displays the baseline control approach selected for the XNR2000. One possible option for providing redundant reactor shutdown control would be the insertion of Re rods inside the radial support tubes shown in the medial plane cross-section. The rods could be included in the design to prevent inadvertent reactor excursions during transportation, pre-launch, or booster transfer. The Re rods would provide an independent back-up safety mechanism but would not be used for reactor control.

CRITICALITY CONFIRMED BY B&W AND ANL



PRATT & WHITNEY XNR2000 CERMET NTRE

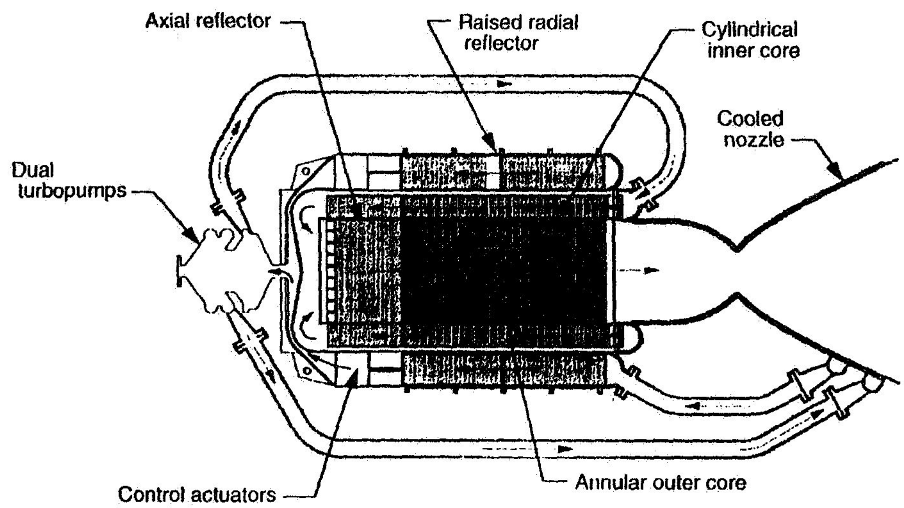
2507/5

XNR2000 Criticality Confirmed by B&W and ANL

The calculated values of system Keff by Pratt & Whitney agree with calculations of Keff conducted independently by Babcock & Wilcox (B&W) and Argonne National Lab (ANL). Pratt & Whitney used both a 16 energy group combine/venture diffusion code analysis and MCNP statistical code analysis to calculate Keff and ANL used MCNP statistical code analysis procedures to calculate Keff.

The plot of Be radial reflector thickness vs. Keff for the baseline configuration, displays the large worth of reactivity for the reflector under approximately 30 cm. This curve indicates that the system can be controlled with neutron reflection. The baseline system employs an 18 cm radial Be reflector and a 20 cm BeO axial reflector.

XNR2000 SYSTEM CAN BE CONTROLLED BY RAISING REFLECTOR SECTION



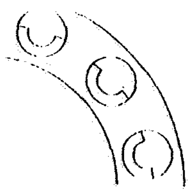
PRATT & WHITNEY XNR2000 CERMET NTRE

25447

<u>XNR2000 System Can Be Controlled By Raising Reflector Sections</u>

The baseline control approach was designed to provide robust reactor control with minimum complexity and weight. The control of the reactor is accomplished by varying the neutron-leakage rate by means of 10{SYMBOL 176 \f "Symbol"} moveable annular segments of the radial reflector. The lower half of each segment is stationary while the upper half translates axially to provide reactor control through the "opening of windows". Nuclear control is provided by 1 bank of 3 segments while fast-shutdown capability is provided by the other, independent, bank of 3 segments. The selected control approach provides the most reactivity worth for the selected reflector size, thus maximizing thrust to weight. The reflector segments are driven by pneumatic piston-type drive mechanisms which provide linear actuation.

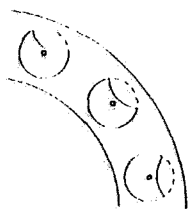
ALTERNATE CONTROL OPTIONS ARE VIABLE



Conventional drums

Estimated worth

$$\frac{\Delta K}{K}\Big|_{TOTAL} = -4\%$$



Rotating windows

$$\frac{\Delta K}{K}\Big|_{TOTAL} = -5\%$$

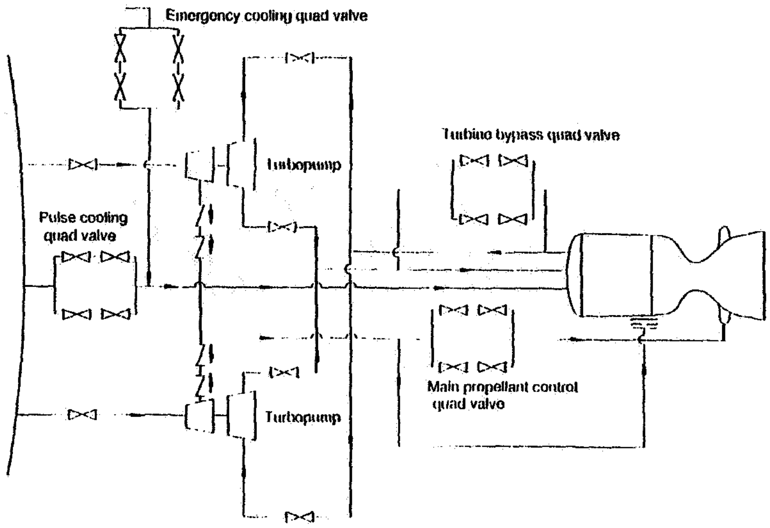
PRATT & WHITNEY XNR2000 CERMET NTRF

25377

Alternate Control Options Are Viable

The baseline control approach was selected for simplicity and reduced weight, however other control options are viable for the XNR2000. Shown here are two such approaches with a preliminary calculation of reactivity worth. Contemporary control drums consisting of Be with partial segments of Re poison material could provide sufficient negative reactivity insertion for control. Additionally, the use of rotating drums with segments of void could be used to provide control through neutron leakage in a rotating drum configuration. The optimum control of the XNR2000 could be achieved through the combination of any of the three approaches presented, providing nuclear control with maximum redundancy.

BASELINE PROPELLANT SYSTEM CONFIGURED WITH DUAL TURBOPUMPS



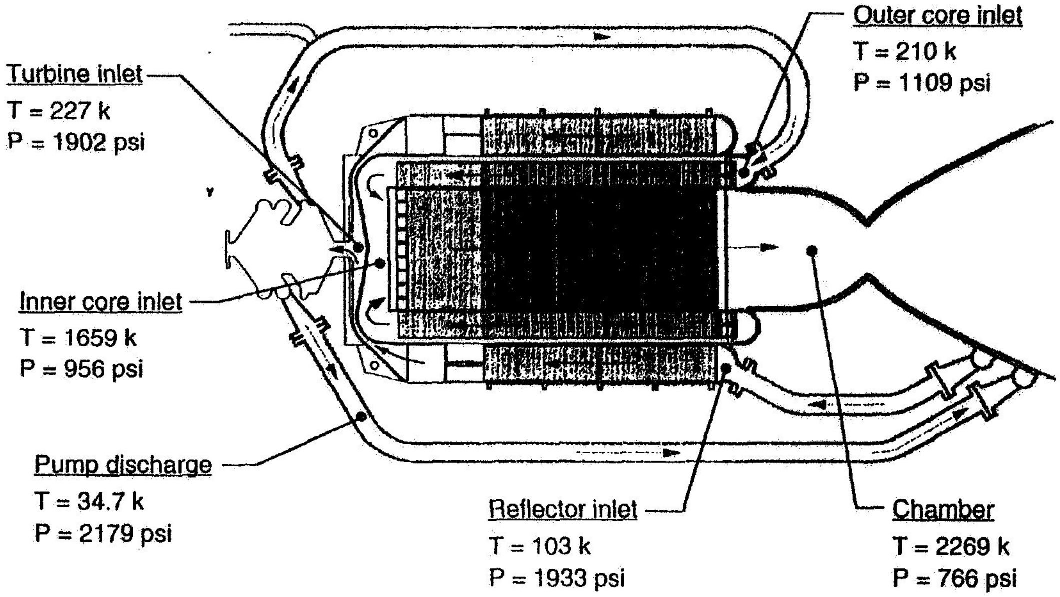
PRATT & WHITNEY XNR2000 CERMET NTRE

25378

Baseline Propellant System Configured With Dual Turbopumps

A flow schematic of the baseline engine is shown. Dual turbopumps are employed with quad valve arrangements to maximize system reliability. Each turbopump delivers 50% of the total reactor flow and can be isolated with block valves in case of a pump out condition. The quad valves consist of 2 block valves followed by 2 control valves arranged in parallel. The 2 pumps pressurize and deliver hydrogen to the nozzle coolant tubes and reflector. The heated hydrogen is then expanded through the turbine and delivered to the reactor. Preliminary investigations indicate that the system could operate at 75% thrust during an engine out scenario. After engine operation pulse cooling of the reactor is provided with pressurized or tank head hydrogen through the pulse cooling quad-valve to remove residual heat generation. An emergency pressurized hydrogen tank would provide pressurized hydrogen to the reactor under a 2 pump out, reactor critical condition.

XNR2000 EXPANDER CYCLE IS ROBUST AND EFFICIENT



PRATT & WHITNEY XNR2000 CERMET NTRE

25448

CERMET ADVANTAGES ESTABLISHED IN GE710/ANL PROGRAMS

Characteristic	Payoff
Demonstrated fabrication	Reduced risk
Fuel matrix / cladding / hydrogen compatibility	Life, FFP retention
High strength and conductivity	Thrust–to–weight, robustness
High temperature operating capability	Specific impulse

Characteristics confirmed by B&W

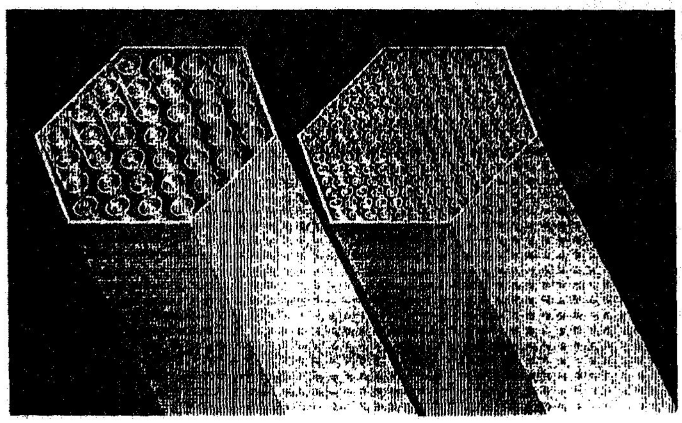
PRATT & WHITNEY XNR2000 CERMET NTRE

25382

Cermet Advantages Established in Ge710/ANL Programs

The XNR2000 builds upon the experience and database of Cermet fuels obtained in the GE710 and ANL programs. The fast spectrum Cermet fuel form was selected to meet the engine requirements of ALARA fuel and fission product release, multiple restart capability and subcriticality under credible accident scenarios. During the GE710/ANL programs the Cermet fuel form displayed tolerance to excessive temperature/power ramps due to the high strength and conductivity of the refractory metal matrix. Additionally Cermet fuel display complete compatibility in the expected hot H2 operating environment as well as cladding and fuel matrix CTE compatibility. Finally the XNR2000 is based upon a fuel form that was successfully fabricated and tested.

PROPOSED FUEL ELEMENT CONFIGURATION HAS BEEN UPDATED



PRATT & WHITNEY XNR2000 CERMET NTRE 25410

Proposed Fuel Element Configuration Has Been Updated

Subsequent to the Mid term evaluation of this study, the baseline concept has been upgraded to incorporate a fuel element based on demonstrated technologies. The baseline fuel form incorporates 37 large diameter coolant channels compared to 169 small diameter coolant channels initially considered for this concept. The max. operating fuel temperature was maintained at 2880K, well within the experimental database. Because of the increased thermal path, fuel centerline to coolant channel surface, between the fuel forms the reactor exit propellant temperature was reduced to 2669K from 2850K. This chamber temperature provides an Isp level of 900 seconds with life greatly in excess of the NASA requirements.

BASELINE FUEL ELEMENT WITHIN FABRICATION EXPERIENCE

Parameter	Inner core element	Outer core element
Number of fuel elements	61	90
Distance across flats [a]	1.40 in.	1.40 in.
Diameter of flow hole [D]	.14 in.	.14 in.
Flow hole pitch	0.215 in.	0.215 in.
Thickness of flow tube [t]	0.007 in.	0.007 in.
Thickness of external can [W]	0.02 in.	0.02 in.
Number of flow holes	37	37
Fuel matrix materials	UO_2–W–Gd_2O_3	UO_2–Mo-Gd_2O_3
Metal in fuel matrix	W	Mo
Fuel can materials	W–Re	Mo–Re
Flow tube material	W	Mo
Uranium enrichment (wt%)	93.0	93.0
Vol. fraction of UO_2 in fuel matrix	0.6	0.6
Vol. fraction of Gd_2O_3 in fuel matrix	0.06	0.06
Vol. fraction of metal in fuel matrix	0.34	0.34
Vol. fraction of Re in external can	0.25	0.50
Vol. fraction of metal in external can	0.75	0.50
Flow hole void fraction	0.3425	0.3425
Total core power (MW)	510	
Total core volume (L)	101.1	
Active core volume (L)	66.47	
Heat transfer area/total flow area	1447.3	
Fuel element height	24 in.	
Radial reflector (Be) thickness	7.1 in.	
Axial top reflector (BeO) thickness	7.87 in.	
W sheet thickness on inner core bottom	1 in.	

PRATT & WHITNEY XNR2000 CERMET NTRE

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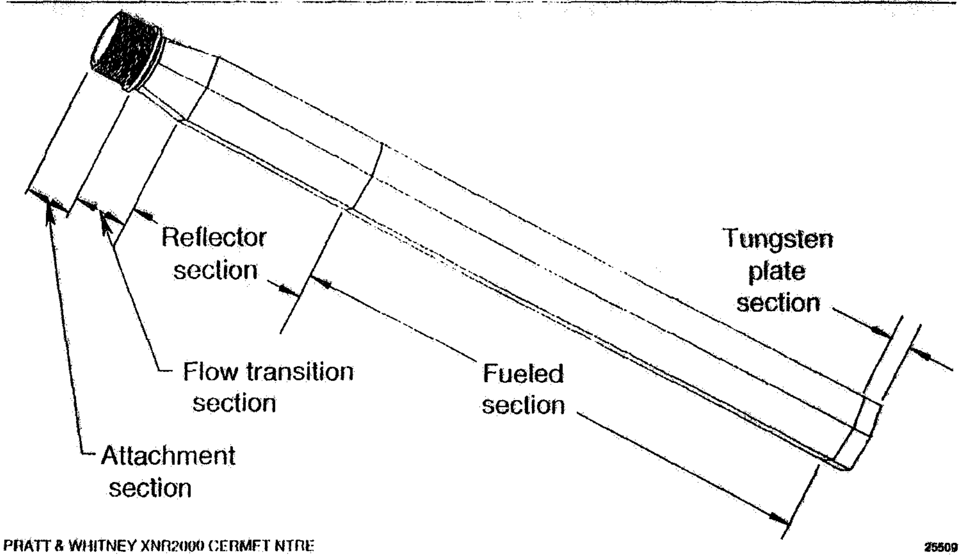
Baseline Fuel Elements Within Fabrication Experience

The selected baseline prismatic Cermet fuel element is based on demonstrated technology. The outer core fuel elements consist of 60 vol%-UO2-Mo fuel matrix contained within Mo-50% Re external core. The inner core fuel elements consists of 60 vol% UO2-W fuel matrix contained within a W-26% Re external core. Rhenium has been incorporated into the external can designs to decrease the ductible-to-brittle transition temperature and provide adequate ductility for cyclic life requirements. All fuel elements have a hexagonal cross section with a 1.4 inch flat-to-flat distance and contain 37 coolant channels .14 inch in diameter.

The coolant channels are coated with the refractory metal contained within the matrix. UO2 is stabilized with 6% GdO3 in both cores to provide fuel stabilization and prevent fuel migration. Fuels elements of this type were successfully fabricated and tested in the early 70's with technology that can be easily recovered and enhanced with a core recent fabricated techniques.

AXIAL REFLECTORS INTERGRAL WITH FUEL ELEMENTS



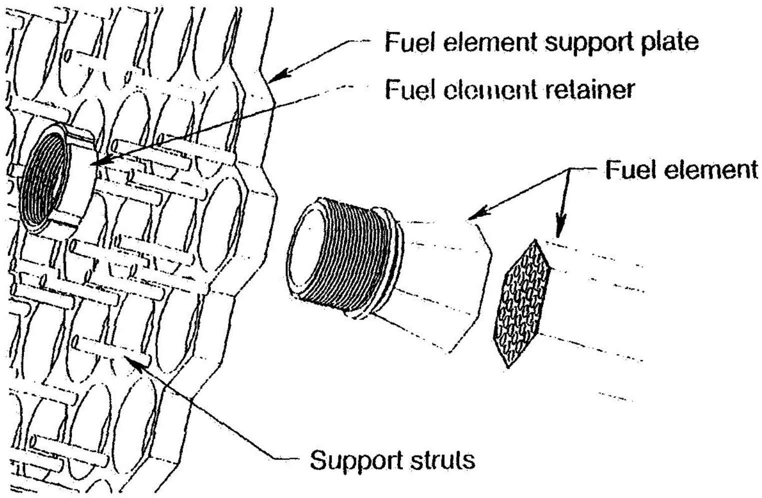
PRATT & WHITNEY XNR2000 CERMET NTRE

25509

<u>Axial Reflectors Integral With Fuel Elements</u>

The baseline prismatic Cermet fuel element for the inner core is shown. The axial BeO reflector section is integral with the fuel element, contained within the same structural support external can. The attachment section is used in a support system. The loaded section of the fuel element is 24 inches (61 cm.) in length and the axial reflector is 7.9 inches (20 cm.).

HIGH STRENGTH FUEL ELEMENTS ALLOW SIMPLIFIED CORE SUPPORT



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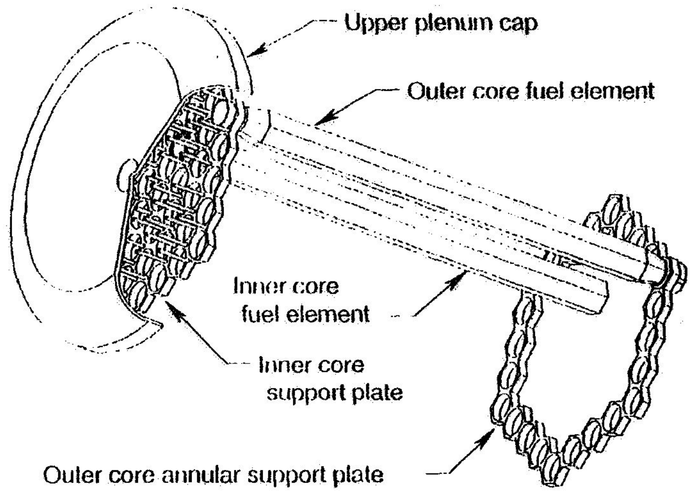
25804

<u>High Strength Fuel Elements Allow Supplied Core Support</u>

The XNR2000 is not susceptible to material neutronic poisoning because of the fast spectrum operation of the reactor. Therefore, high strength refractory metals can be used in both the fuel matrix and support structure to eliminate the need for tie rods. The baseline conceptual core support design is shown below. The fuel elements are simply supported, at the hydrogen inlet end, to the support plate with a threaded fuel element retainer. The fuel elements are placed in tension because of propellant friction and accelerational pressure drop which acts to increase the natural frequency of the fuel element and reduce the propensity for flow induced vibration.

CONCEPTUAL CORE ASSEMBLY APPROACH



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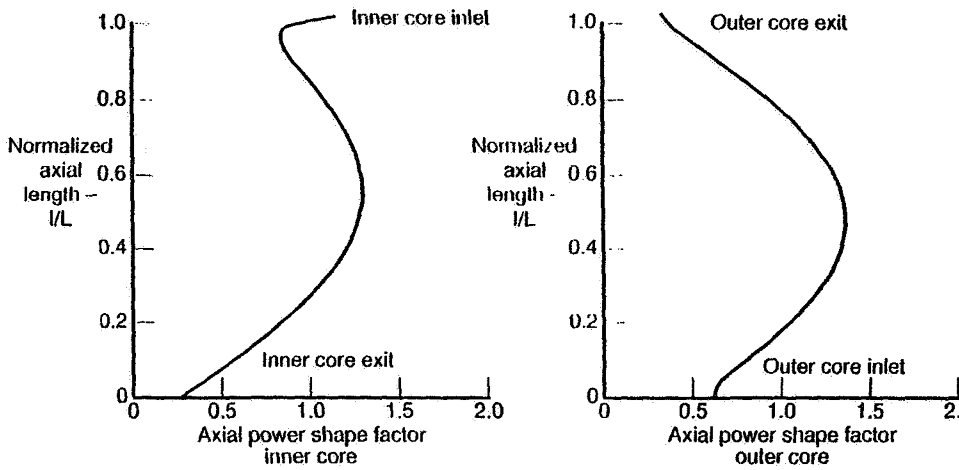
25806

Conceptual Core Assembly Approach

Shown below is the conceptual inner and outer core fuel element support approach. The outer core elements are simply supported at their cold end by a lower grid plate which is bolted to the inner pressure vessel. The outer core elements are allowed to translate through the upper support plate to allow for axial thermal growth. The inner core fuel elements are rigidly attached at their cold end by the upper grid plate. The upper grid support plate is bolted to the inner pressure vessel with additional support provided by axial struts attached to the upper plenum head.

A tungsten shroud will be used between the two cores to act as a thermal baffle and provide a compressive spring preload against radial inner core fuel element growth. The tungsten shroud will conform to the hexagonal cross-section of the fuel elements and extend from the upper support plate to the nozzle chamber. The shroud will transition from a hexagonal cross-section to a circular cross-section in the chamber region. The nozzle coolant tubes will run behind this shroud in a circular pattern to provide chamber cooling.

AXIAL POWER DISTRIBUTION PREDICTED FOR THERMAL HYRAULIC ANALYSIS



PRATT & WHITNEY XNR2000 CERMET NTRE

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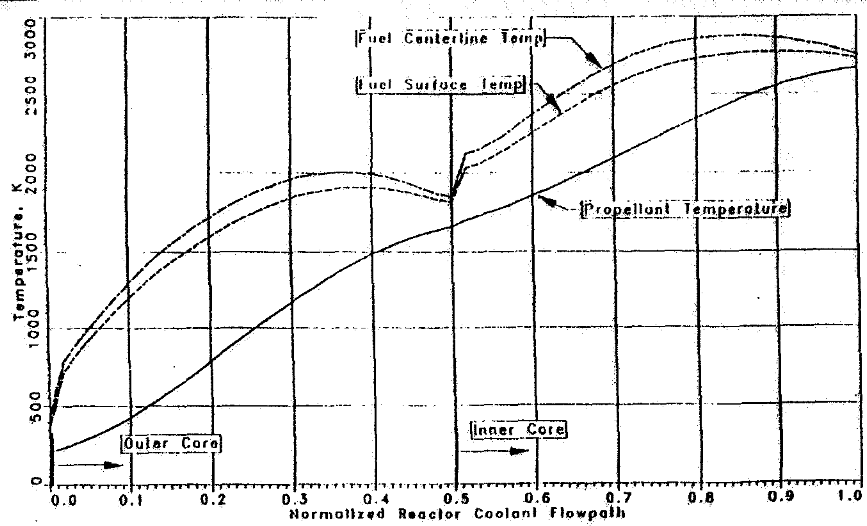
<u>Axial Power Distribution Predicted For Thermal Hydraulic Analysis</u>

The predicted average axial power shape factors for the inner and outer cores are shown. These power profiles were determined using the 3-D diffusion theory code, BOLD VENTURE, and benchmarked with MCNP statistical codes. The inner core power profile decreases at the exit of the reactor where the temperatures are the highest. The sharp increase in power at the inner core inlet is caused by the BeO axial reflector located directly above the reactor. These power profiles were determined to conduct a coupled neutronic/thermal hydraulic analysis of the XNR2000 reactor.

PEAK FUEL TEMPERATURE IS MAINTAINED BELOW 2900K



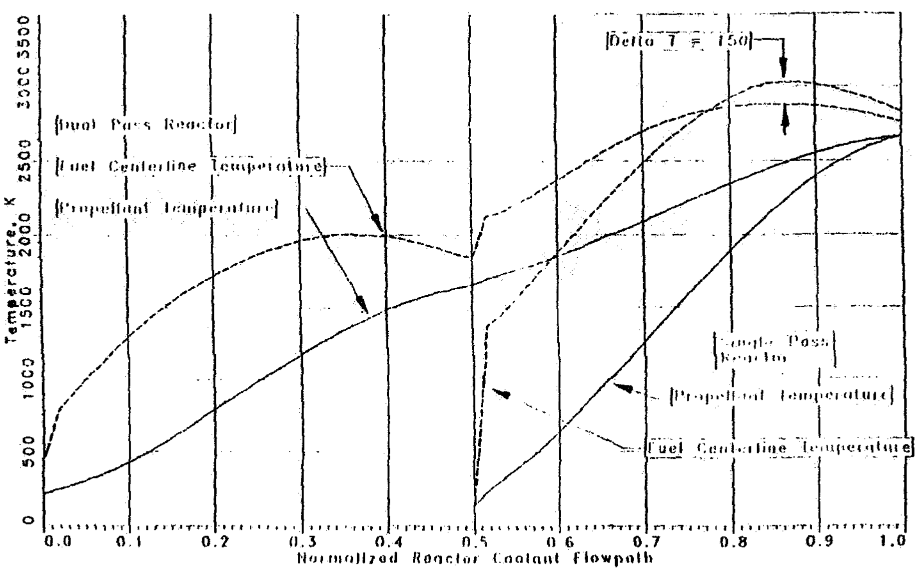
PRATT & WHITNEY XNR2000 CERMET NTRE

25389

<u>Peak Fuel Temperature Is Maintained Below 2900k</u>

The calculated propellant, fuel surface, and fuel centerline temperature distribution within the XNR2000 reactor at full power operating condition is shown. The temperature distribution is plotted against the normalized reactor coolant flowpath location, where 0.0 corresponds to the outer core inlet and 1.0 corresponds to the inner core exit. This temperature distribution was calculated using a one-dimensional coupled thermal hydraulic/neutronic analysis benchmarked with detailed 3-dimensional computational fluid dynamics, CFD, procedures. As shown in the figure, the maximum fuel temperature reached in the inner core is 2880K and 2000K in the inner core. These maximum fuel temperatures were selected for design operation to exceed life requirements and assure positive fission product and fuel retention. A propellant chamber temperature of 2669k was calculated using a 2880k max fuel temperature as the upper limit.

DUAL PASS REDUCES FUEL TEMPERATURES AND AXIAL THERMAL GRADIENTS



Dual Pass Reduces Fuel Temperatures And Axial Thermal Gradients

The figure displays several benefits of the dual pass reactor flow configuration. The dual pass provides reduced axial thermal gradients in the fuel elements. As shown in the figure, a temperature gradient of 1500K appears across the outer core elements and a gradient of 1100K appears across the inner core elements, in the dual pass configuration. However, in a single pass configuration a temperature gradient of 2600K appears across each fuel element. The dual pass flowpath reduces the axial thermal gradients of the elements by approximately 50%, reducing thermal stresses and increasing fuel tolerance to power cycling. Additionally, in a dual pass reactor max fuel temperatures are reduced by approximately 160(SYMBOL 176 \f "Symbol")K for equal propellant chamber temperatures and power density. This is a result of increased heat flux and decreased convective heat transfer in the single pass configuration, for equivalent reactor power density levels.

C-3

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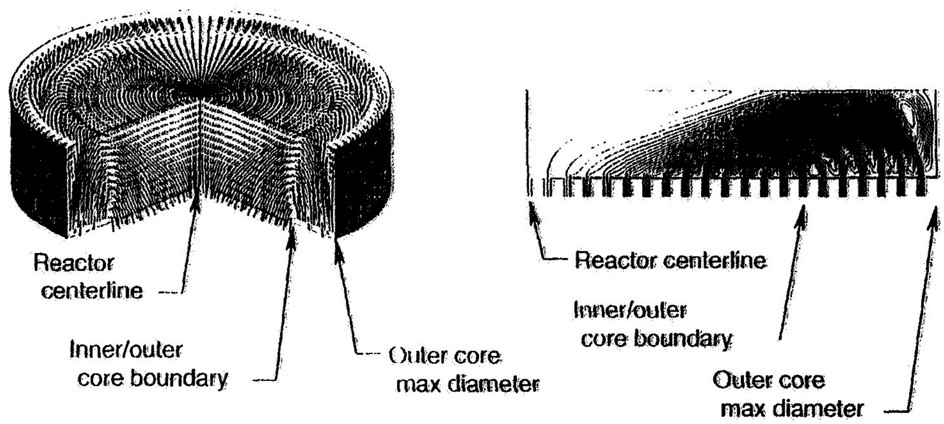
PRELIMINARY UPPER PLENUM CFD RESULTS

Stream function of jet–induced flow



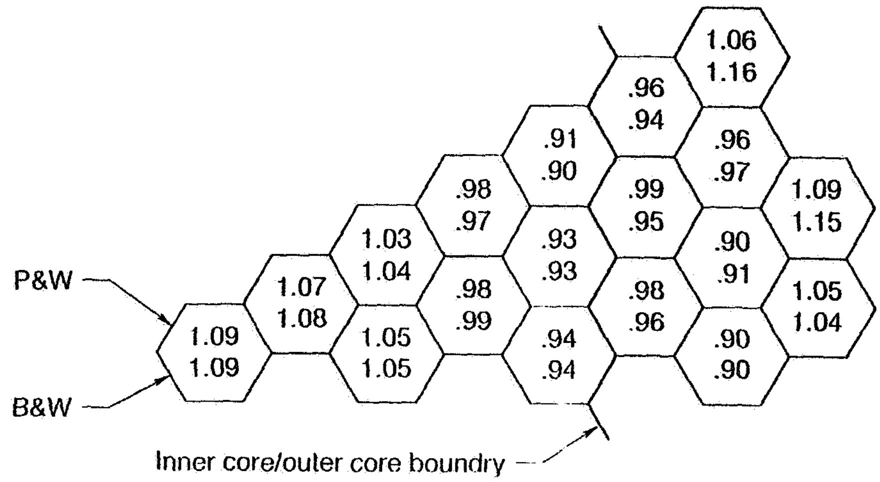
PRATT & WHITNEY XNR2000 CERMET NTRE

25385

<u>Preliminary Upper Plenum CFD Results</u>

A Computational Fluid Dynamic (CFD) analysis was conducted to evaluate the flow distribution and heat transfer in the XNR2000 reactor coolant channels and upper plenum region. The predicted flow distribution in the upper plenum is shown below. The results of the CFD analysis were used in the upper plenum design and to benchmark the one dimensional thermal hydraulic reactor analyses.

XNR2000 RADIAL POWER DISTRIBUTION CONFIRMED BY B&W



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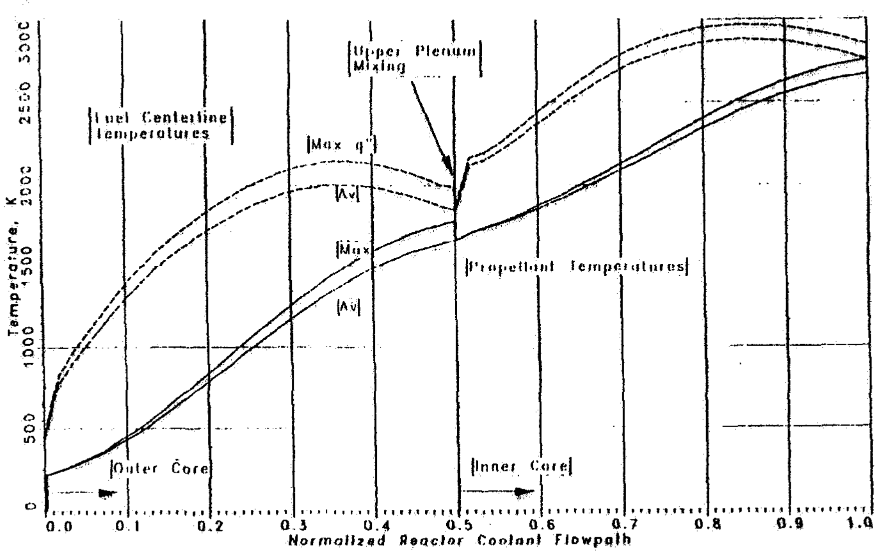
<u>XNR2000 Radial Power Distribution Confirmed By B&W</u>

The calculated rodwise normalized power distribution within a segment of symmetry of the XNR2000 reactor is shown. Close agreement between the calculated results of Pratt & Whitney and Babcock & Wilcox is shown. As expected, the maximum power peak of the inner core appears at the center of the reactor while the power peak of the outer core appears closest to the radial Be reflector. These results were used in the thermal hydraulic analysis to conduct power/flow matching evaluations. As shown in the diagram the maximum peak-to-average fuel element power level was calculated to be 1.09 for both the inner and outer cores.

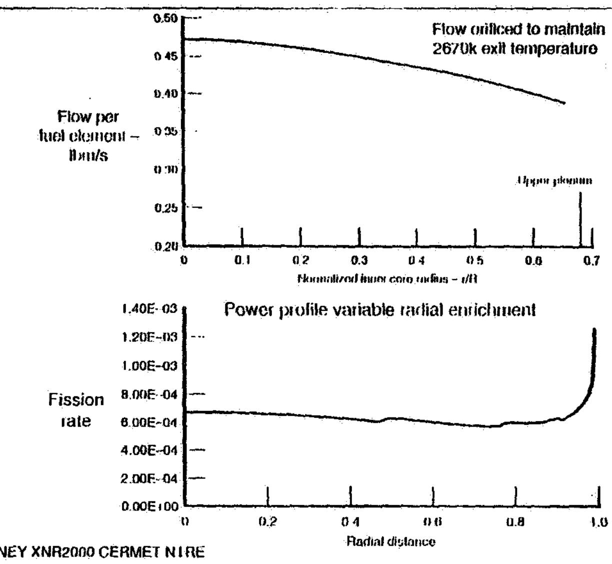
UPPER PLENUM MIXING FLATTENS RADIAL POWER PROFILE



<u>Upper Plenum Mixing Flattens Radial Power Profile</u>

The calculated propellant and fuel centerline temperature distribution within the XNR2000 reactor for fuel elements having the average and maximum peak-to-average power levels is shown. The calculated radial power distribution shown on the previous chart, was used to conduct a thermal hydraulic evaluation of the reactor to determine the impact of peak power levels on reactor temperatures. The analysis displays the upper plenum mixing advantage of the dual-pass core. Any outer core hot channeling effect due to uneven power profiles is removed from the inner core because of thermal momentum fluid mixing in the upper plenum. This mixing reduces the propellant and therefore reactor temperatures in the inner core. The energy and momentum mixing allows for up to 15% power peaking in the outer core without orificing. As shown, the maximum temperature is approximately 2950k for the inner core and 2200 for the outer core in the fuel elements having the maximum power levels. This *analysis displays the worst case scenario in which no attempt is made to flatten the power profile.*

PROFILE CAN BE ADDRESSED BY VARIABLE ENRICHMENT OR ORIFICING



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Profile Can Be Addressed By Variable Enrichment Or Orificing

Two methods of addressing the power profile were evaluated and both were found to be acceptable. The first approach of handling the variable power profile was orificing the propellant flow in the inner core to provide a constant 2670K reactor exit temperature. By orificing the flow at the inlet of each fuel element the proper flow rate can be delivered to each element depending on the element power level. Shown below is the fuel element flowrate, as a function of inner core radius, required to provide a constant reactor exit temperature.

The second possible approach to flatten the power profile evaluated was variable radial Uranium enrichment. The enrichment within both the inner and outer cores was varied to determine the impact on radial power distribution. As shown in the figure a nearly constant power profile was obtained by varying the enrichment by approximately 4% across the reactor radius.

DUAL PASS CONFIGURATION HAS SIGNIFICANT ADVANTAGES

- Flat radial power profile
- Positive flow/power matching
- Upper plenum mixing reduces peak temperature
- High temperature inner core isolation
- Reduced element axial thermal gradient

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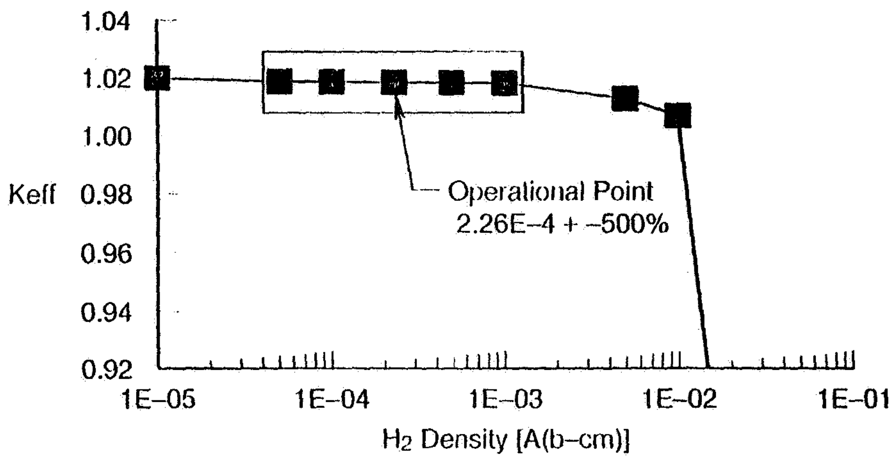
<u>Dual Pass Configuration Has Significant Advantages</u>

The primary attractive features provided by the dual pass reactor core are summarized. A flat radial power profile is provided by the dual-pass reactor due to the averaging of power distributions relative to two distinct regions. Positive flow/power matching is achievable because of the separation of the inner and outer cores. The maximum fuel element power shape factors appear in the outer core region because of the proximity of the radial reflector. However, because the outer core serves as the first pass , the coolest hydrogen propellant passes through the outer core and eliminates fuel temperature concern. Additionally, upper plenum mixing of the hydrogen serves to eliminate the outer core power peaks from the inner core fuel elements. The dual pass configuration isolates the hot inner core fuel elements from the rest of the engine system. This isolation provides material flexibility allowing the use of lighter weight Moly based fuel elements in the outer core and a Be radial reflector which provides the most reactivity worth for the weight. The most obvious benefit of the dual pass core is the reduced axial thermal gradients and consequently thermal stress loads placed on the fuel elements.

COMPLICATIONS OF H_2 MODERATION ELIMINATED DURING STARTUP, SHUTDOWN, AND THROTTLING



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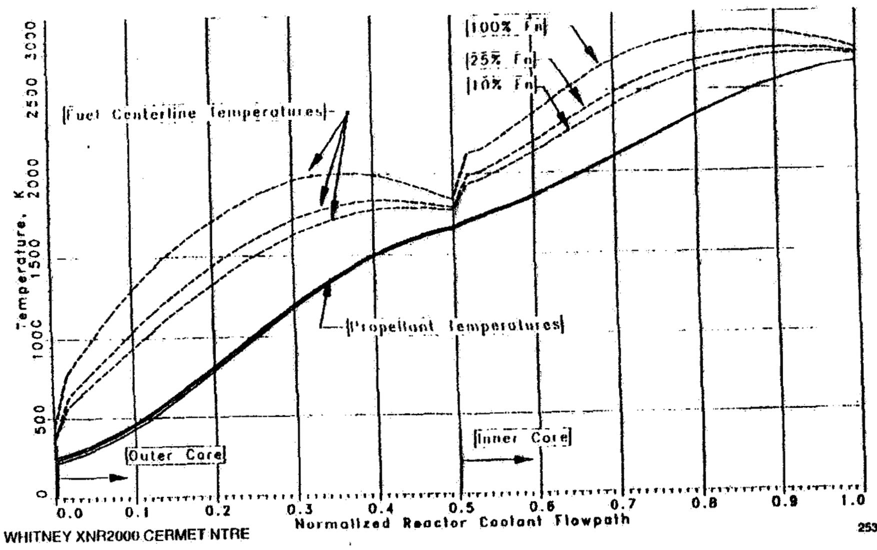
Complications Of H2 Moderation Eliminated During Startup, Shutdown, and Throttling

There is no impact of Hydrogen moderation on the fast spectrum XNR2000 reactor. The calculated effect of Hydrogen density on system K_{eff} is shown. The complications of reactivity feedback from the hydrogen propellant and potential for thermal instability is eliminated during transient and steady-state operation in the KNR2000.

PEAK FUEL TEMPERATURE DECREASES AT THROTTLED CONDITIONS



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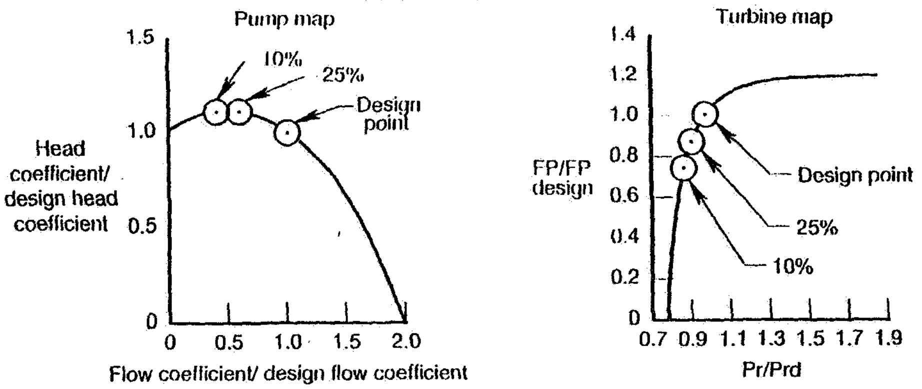
Peak Fuel Temperature Decreases At Throttled Conditions

The calculated propellant and fuel centerline temperatures are shown for the baseline XNR2000 at full thrust, 25% thrust and 10% thrust throttled conditions. As displayed in the chart the peak fuel temperatures within the reactor decrease as the engine is throttled. The reduced reactor temperatures result from the reduced power flux required to deliver the throttled mass flow rate to the design point temperature levels. This quasi steady analysis was simplified because of the negligible effect of H2 moderation on the reactivity of the core.

STARTUP, SHUTDOWN AND THROTTLING, UNAFFECTED BY H_2 MODERATION

Configuration allows throttling to 10% thrust at design ISP



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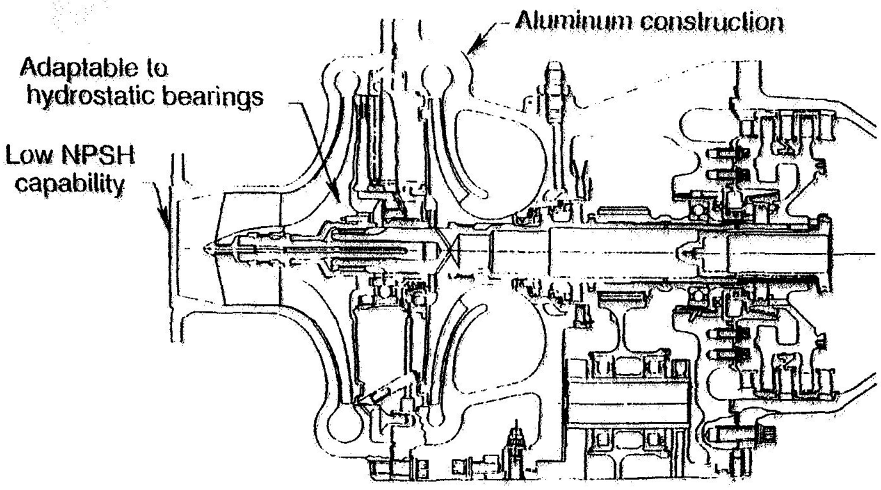
Startup, Shutdown And Throttling, Unaffected By Hydrogen Moderation

The pump and turbine operating map of the XNR2000 is shown for throttled and design point conditions. The RL10 upper stage expander cycle rocket engine turbopump characteristics were assumed in this analysis. This analysis indicates that the configuration allows throttling to at least 10% thrust at design specific impulse.

INDIVIDUAL TURBOPUMP REQUIREMENTS ARE SIMILAR TO RL10



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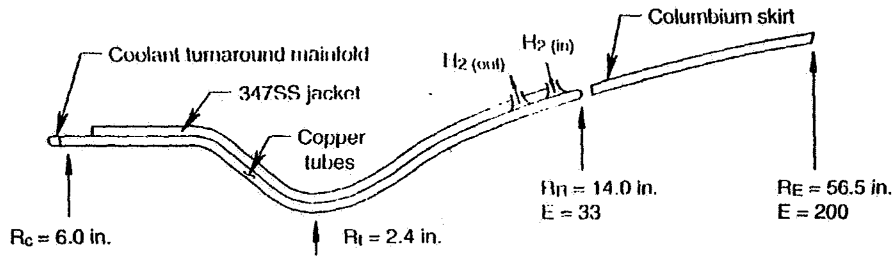
Individual Turbopumps Requirements Are Similar to RL10

Demonstrated characteristics of the RL10 engine turbopumps will be required of turbopumps used in an NTRE for manned space exploration missions. The XNR2000 conceptual design employs a two stage centrifugal pump that is similar in flow rate and head rise to the RL10 turbopump, driven by a turbine operating at cool inlet temperatures. The system requirements call for throttling to at least 25% thrust at rated temperature and operation at low NPSH levels. These requirements are similar to those of the RL10 liquid hydrogen turbopumps. The RL10 turbopumps deliver pressurized hydrogen to the RL10 engine for upper stage applications. This pump has successfully demonstrated zero to low NPSH capability and throttling down to 2% flow. With the incorporation of *hydrostatic bearings, operation in a radiation environment can be achieved because of the aluminum construction.* For these reasons the characteristics of the RL10 Turbopumps were used in the study of The XNR2000 concept, and that a scaled or derivative version of this proven pump would be employed in the design.

NOZZLE IS ACTIVELY COOLED COPPER WITH AN UNCOOLED SKIRT



Regen section		**Skirt**	
Coolant configuration	Two pass	Coolant configuration	Radiation
Number of tubes	300	Skirt material	Columbium
Tube material	Glidcop	Max heat flux	1 Btu/in^{2}sec
Max heat flux	51 Btu/in^{2}sec	Max skirt temperature	1792K (2766°F)
Max tube temperature	811K (1000°F)		
Pressure drop	225 psi		

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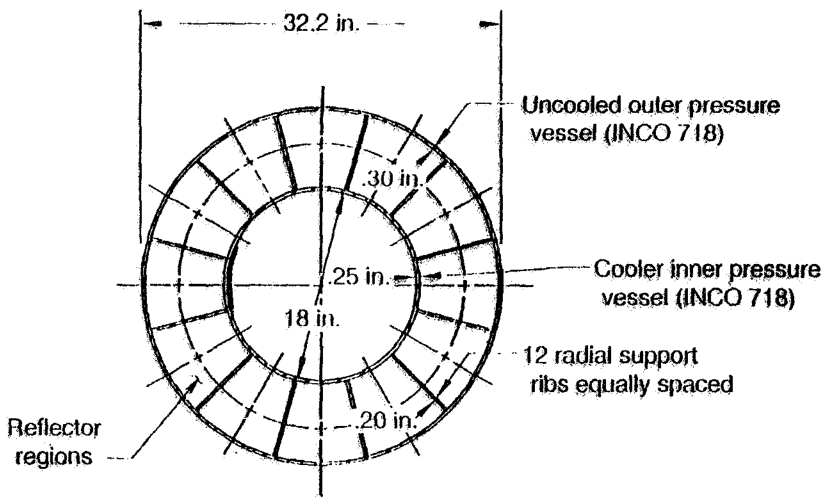
<u>Nozzle Is Actively Cooled Copper With An Uncooled Skirt</u>

The XNR2000 employs a regeneratively cooled chamber and nozzle and radiativly cooled nozzle skirt. The nozzle and chamber is cooled to an area ratio of 33 with 300 copper tubes in a two pass configuration with 30% of the total engine flow. The chamber pressure vessel consists of a 347 Stainless Steel jacket surrounding the copper tubes. The system employs a Colombian nozzle skirt from an area ratio of 33 to 200 which is radiativly cooled.

XN2000 PRESSURE VESSEL IS SIMILAR TO ANL APPROACH



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<u>XNR2000 Pressure Vessel Is Similar To ANL Approach</u>

The XNR2000 employs an outer uncooled pressure vessel which surrounds the radial reflector and a regeneratively cooled inner pressure vessel which surrounds the reactor. The pressure vessel material considered is Inconnel INCO 718. Because the inner pressure vessel is subjected to a collapsing pressure of approximately 800 psi, longitudinal radial support ribs would be employed to transmit this load to the outer vessel. The radial support ribs would serve to separate and house the annular reflector segments. The two pressure vessels are capped at the top of the reactor by hemispherical heads. Hydrogen exits the reflector region and flows between the primary and secondary heads to cool the primary head covering the inner pressure vessel and provide additional heat input to the turbine.

XNR2000 BASELINE DESIGN EXCEEDS NASA REQUIREMENTS

	Baseline
Thrust (lb)	25,000
Isp (sec)	900
T/W	5.3
Reactor power (Mw)	510
Power density (Mw/L)	9.4
Max fuel temp (K)	2,880
Chamber temp (K)	2,669
Chamber pressure (psia)	766
Total flow (lb/sec)	27.8
Pump tip speed (ft/sec)	1,460
Turbine inlet temp (K)	227
Nozzle area ratio	200
Nozzle exit dia (ft)	5.8
Max engine length (ft)	15.3
Stowed engine length (ft)	11.0
No. of inner fuel elements	61
No. of outer fuel elements	90
Throttling at design Isp (%)	10

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<u>XNR2000 Baseline Design Exceeds NASA Requirements</u>

The table displays the cycle performance information of the baseline XNR2000. The baseline XNR2000 delivers 25,000 lb. of thrust at a specific impulse of 900 sec. with a thrust to weight ratio of 5.3. This power balance information was generated using the Marshall Space Flight Center/P&W Rocket Engine Transient Simulation (ROCKETS) System

XNR2000 ENGINE PERFORMANCE

Thrust = 25,000 lbf
T/W = 5.3
Isp = 900.0 sec

PROPELLANT FLOW ENGINE STATION CONDITIONS

Station Location	Pressure (psia)	Temperature (Deg K)	Flow (lbm/s)	Enthalpy (Btu/lbm)	Density (lbm/ft**3)
Engine Inlet	26.7	20.6	14.0	-108.0	4.38
Pump Inlet	25.7	20.6	14.0	-108.0	4.38
Pump Exit	2179.3	34.7	14.0	13.0	4.56
Nozzle Coolant Inlet	2157.6	34.8	8.4	13.0	4.55
Reflector Coolant Inlet	1932.6	103.1	28.1	440.9	1.77
Turbine Inlet	1901.6	226.9	11.8	1343.7	0.80
Turbine Exit	1218.2	207.2	11.8	1199.9	0.58
Outer Core Inlet	1108.9	210.4	27.8	1221.6	0.52
Inner Core Inlet	956.3	1659.4	27.8	8865.0	0.06
Chamber	765.9	2668.7	27.8	18188.3	0.03

REACTOR CHARACTERISTICS

Two-Pass Design		
Inner Core Diameter	11.5	in
Outer Core Diameter	18.1	in
Reflector Diameter	32.2	in
Pressure Drop	344.1	psia
Max. RX Fuel Temp.	2880.0	K
Outer Core Fuel Mt'l	Mo-UO2,90	
Inner Core Fuel Mt'l	W-UO2,61	
Power Density	9.41	MW/l
Total Power	510.4	MW

NOZZLE CHARACTERISTICS

Nozzle Area Ratio	200.	
Throat Area	18.8	in**2
Exit Dia.	5.8	ft
Nozzle C*	16443	ft/s
Nozzle Length	10.6	ft
Total S.A.	22524	in**2
Regen. Construction	Cu Tubes	
Rad. Construction	Cb Sheet	

PUMP CHARACTERISTICS

Overall Efficiency	73.2	%
Head Rise	69,018	ft
NPSH Avail.	302.9	ft
Speed	71,323	RPM
Power	2403.2	HP
Vol. Flow Rate	1379	gpm
Stg I Flow Coeff.	0.114	-
Stg II Flow Coeff.	0.113	-
Stg I Head Coeff.	0.521	-
Stg II Head Coeff.	0.521	-
Utip 1	1460.	ft/s
Utip 2	1460.	ft/s

TURBINE CHARACTERISTICS

Inlet Temperature	226.9	K
Inlet Pressure	1901.6	psia
Mass Flow	11.8	lbm/s
Overall Efficiency	85.4	%
Speed	71,233	RPM
Pressure Ratio	1.56	-
Inlet Flow Parameter	0.125	-
Overall Velocity Ratio	0.54	-
DH Actual	143.8	Btu/lb
AN**2(E-08)	193.	
Mean Dia.	4.66	in

OPERATION AT 2500K CAN BE ACCOMODATED WITHIN BASELINE CONFIGURATION

	Baseline	
Thrust (lb)	25,000	25,000
Isp (sec)	900	865
T/W	5.3	5.3
Reactor power (Mw)	510	492
Power density (Mw/L)	9.4	9.1
Max fuel temp (K)	2,880	2,740
Chamber temp (K)	2,669	2,500
Chamber pressure (psia)	766	758
Total flow (lb/sec)	27.8	28.9
Pump tip speed (ft/sec)	1,460	1,482
Turbine inlet temp (K)	227	216
Nozzle area ratio	200	200
Nozzle exit dia (ft)	5.8	5.8
Max engine length (ft)	15.3	15.3
Stowed engine length (ft)	11.0	11.0
No. of inner fuel elements	61	61
No. of outer fuel elements	90	90
Throttling at design Isp (%)	10	10

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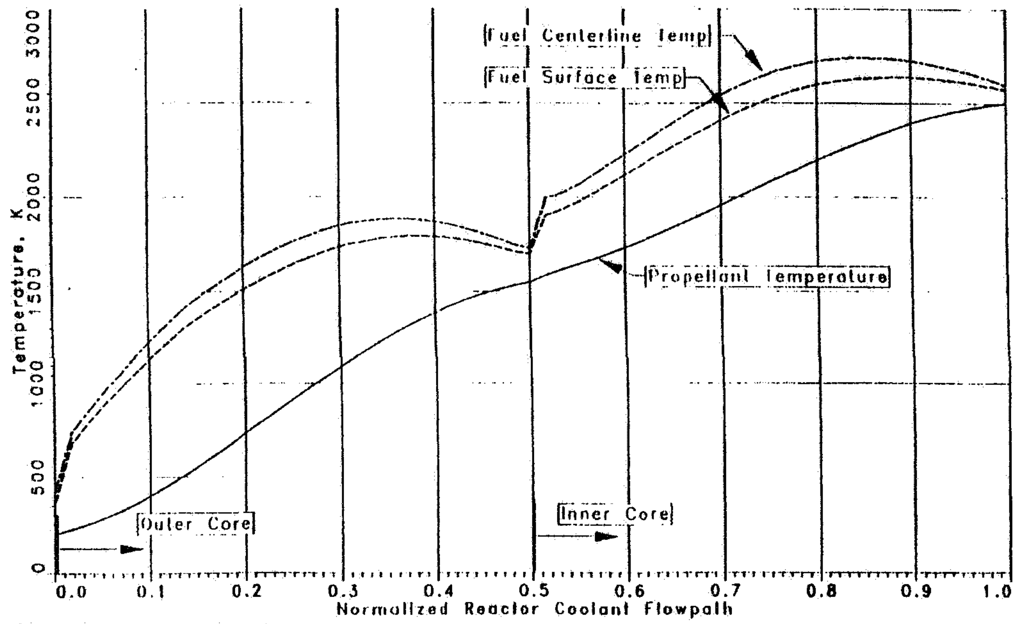
Operation at 2500k Can Be Accommodated Within Baseline Configuration

The baseline cycle information is displayed and compared to the XNR2000 engine operating at a chamber temperature of 2500K. The power balance for both cycle points was generated by requiring the reactor exit Mach numbers to equal 0.3 and deliver 25,000 lb. of thrust.

PEAK FUEL TEMPERATURE DROPS TO 2740K FOR 2500K PROPELLANT DELIVERY



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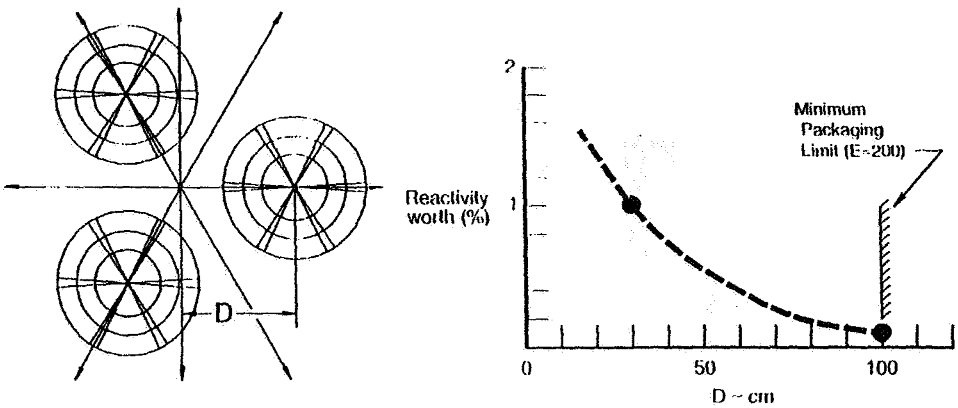
PRELIMINARY ENGINE CLUSTERING STUDY INDICATES LIMITED NEUTRONIC INTERACTION



Three engine clustering arrangement

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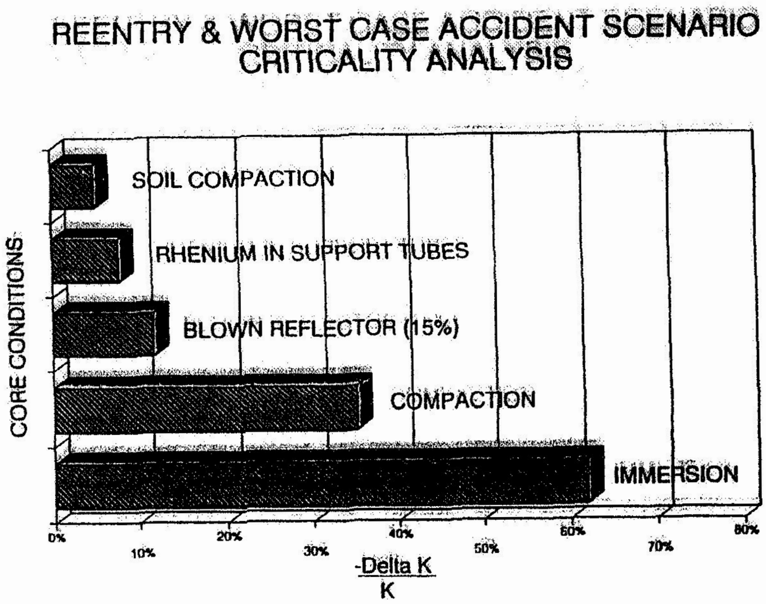
<u>Preliminary Engine Clustering Study Indicates Limited Neutronic Interaction</u>

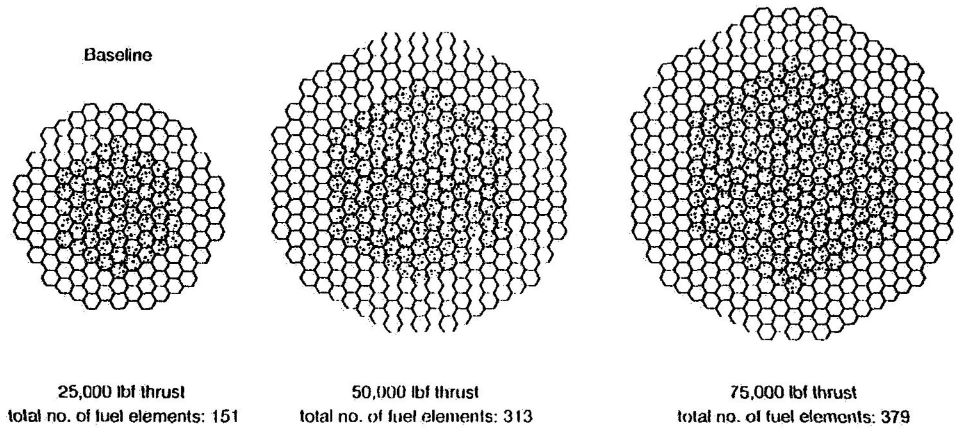
A conservative engine clustering model was developed and the K_{eff} was evaluated for a cluster of three XNR2000 baseline engines as a function of separation distance. The separation distance is defined as shown in the figure. As displayed in the chart, core neutronic coupling was found to have no effect in clustering engines for distances required to account for nozzle skirts.



Reentry & Worst Case Accident Scenario Criticality Analysis

A 16-group diffusion code (VENTURE/COMBINE) analysis was conducted to determine worst case accident scenario criticality. The negative reactivity insertion is shown for several accident scenario core conditions. The XNR2000 would go subcritical for all accident conditions evaluated. The largest negative reactivity insertion occurred for water immersion. The impact of rhenium rods in tubes surrounding the outer core elements was also evaluated and found to provide adequate negative reactivity insertion to be used as a potential back-up safety mechanism. The blown reflector analysis was conducted assuming that 15% of the total reflector was removed from the system.

DESIGN ALLOWS THRUST FLEXIBILITY WITH COMMON FUEL ELEMENTS



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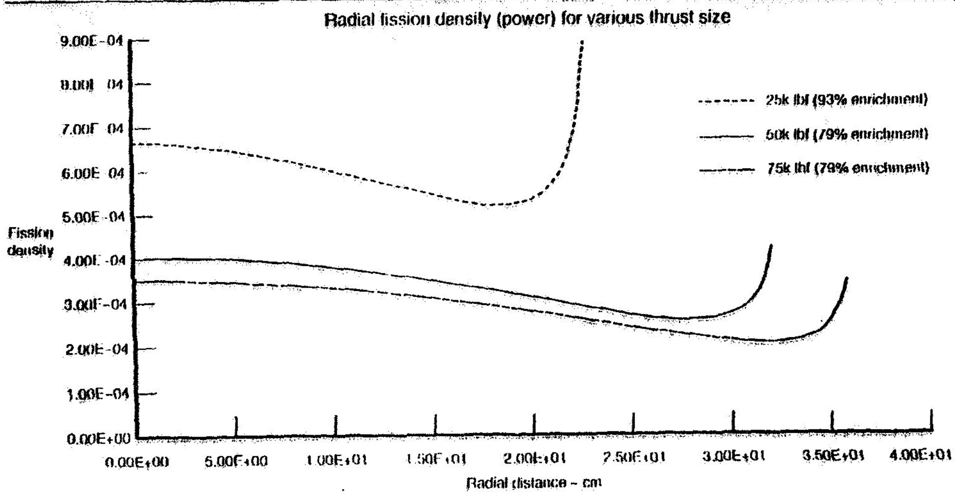
Design Allows Thrust Flexibility With Common Fuel Elements

The XNR2000 was configured to provide thrust flexibility. The system can provide thrust ranging from approximately 20,000 lb to 90,000 using the same fuel element design, core configuration, and support methodology by simply varying the number of inner and outer core fuel elements.

REACTOR NEUTRONICS BEHAVIOR SIMILAR OVER THRUST RANGE



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Reactor Neutronics Behavior Similar Over Thrust Range:

Radial power profiles for three XNR2000 core sizes are shown. The 50,000 lbf and 75,000 lbf can be made critical with 79% enriched fuel at the fuel-metal volume ratio of 60/40.

XNR2000 CYCLE PARAMETERS ARE SIMILAR FOR VARIOUS THRUST SIZES

	Baseline		
Thrust (lb)	25,000	50,000	75,000
Isp (sec)	900	901	897
T/W	5.3	6.6	7.9
Reactor power (Mw)	510	1,022	1,513
Power density (Mw/L)	9.4	9.1	11.1
Max fuel temp (K)	2,880	2,880	2,880
Chamber temp (K)	2,669	2,676	2,657
Chamber pressure (psia)	766	735	836
Total flow (lb/sec)	27.8	55.5	83.6
Pump tip speed (ft/sec)	1,460	1,527	1,738
Turbine inlet temp (K)	227	230	257
Nozzle area ratio	200	200	200
Nozzle exit dia (ft)	5.8	8.3	9.5
Max engine length (ft)	15.3	20.3	22.7
Stowed engine length (ft)	11.0	12.4	12.0
No. of inner fuel elements	61	127	169
No. of outer fuel elements	90	186	210
Throttling at design Isp (%)	10	TBD	TBD

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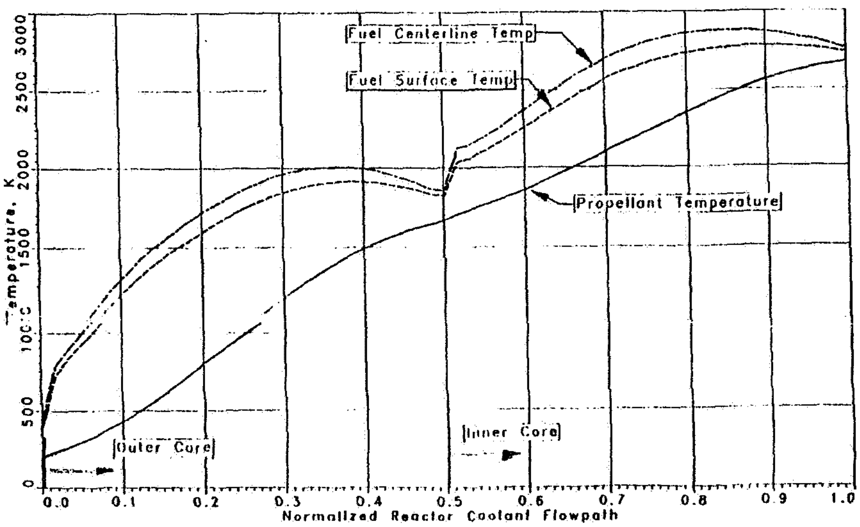
REACTOR THERMAL HYDRAULICS AT 50K ARE SIMILAR TO BASELINE



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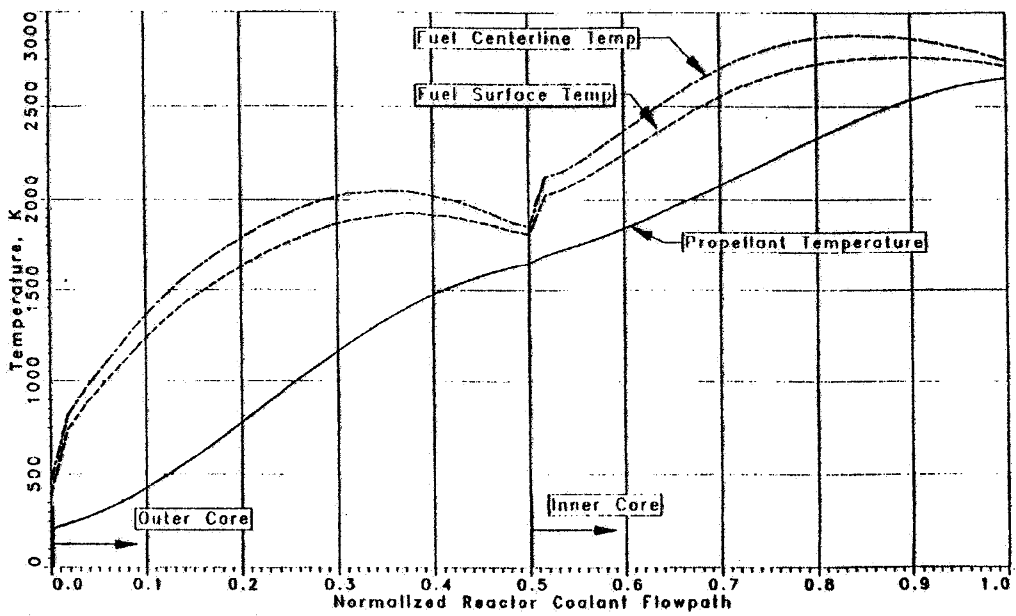
REACTOR THERMAL HYDRAULICS AT 75K THRUST ARE SIMILAR TO BASELINE



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CERMET ENGINE WEIGHT SUMMARY VS THRUST SIZE

Thrust level	25,000 lb	50,000 lb	75,000 lb
Inner core	940	1,662	2,212
Outer core	937	1,644	1,856
Support structure	115	250	425
Internal shield	250	300	310
Axial reflector	50	80	100
Radial reflector and control	500	800	1,000
Valves and controller	425	525	590
Pressure vessel	550	800	1,000
Upper core assembly	220	300	400
Nozzle skirt	250	500	750
Turbopump	75	125	175
Thrust structure	440	600	700
Total engine (lb)	4,752	7,586	9,518
T/W	5.26	6.59	7.88

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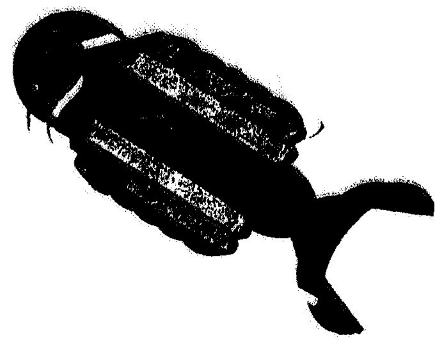
25400

Cermet Engine Weight Summary vs. Thrust Size

A Weight Summary of the XNR2000 for the thrust levels evaluated is shown. The thrust-to-weight ratios for the XNR2000 are high relative to other conventional NTRE's. Several features contribute to the high thrust to weight of the XNR2000. The XNR2000 can operate at a high power density because of the high conductivity of the Cermet fuel and the thermal fluid mixing in the upper plenum. The fast spectrum provides a compact core with no moderator material and the high strength refractory metal fuel elements allow a lightweight support structure. The use of refractory methods and the compact core design reduces required shielding weight. Additionally, the separation of the reactor into two regions allows the use of a lightweight Molybdenum based matrix in the outer core. These features of the XNR2000 NTRE contribute to the high thrust-to-weight.

CERMET APPROACH PROVIDES HIGH PERFORMANCE AND LOW RISK



Thrust = 25,000 lb	Thrust = 50,000 lb	Thrust = 75,000 lb
Isp = 900 sec	Isp = 901 sec	Isp = 897 sec
T/W = 5.3	T/W = 6.6	T/W = 7.9
Dia_{Max} = 5.8 ft	Dia_{Max} = 8.3 ft	Dia_{Max} = 9.5 ft
Stowed length = 11.0 ft	Stowed length = 12.4 ft	Stowed length = 12.0 ft
Deployed length = 15.3 ft	Deployed length = 20.3 ft	Deployed length = 22.7 ft

PRATT & WHITNEY XNR2000 CERMET NTRE

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Cermet Approach Provides High Performance and Low Risk

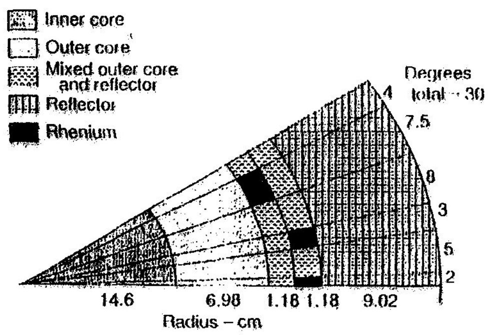
A conceptual NTRE, the XNR2000, has been presented that is powered by a fast spectrum, cermet fueled reactor core. The baseline XNR2000 system delivers 25,000 lbf of thrust at a specific impulse of 900 seconds and thrust to weight of 5.3. The distinguishing features of this system are the dual-pass reactor configuration and fast spectrum, cermet fueled reactor. These features have been incorporated into the design, as well as knowledge gained from the ROVER/NERVA, GE710 and ANL programs, to develop a safe and robust Nuclear Thermal Rocket Engine for manned space exploration missions.

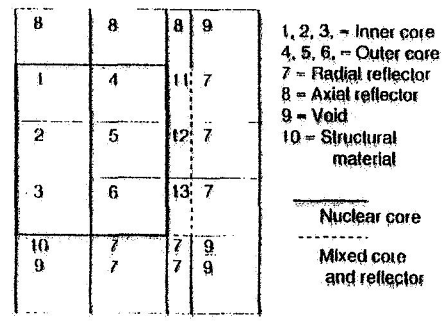
XNR2000 NEUTRONICS ARE BENCHMARKED AND CONFIRMED

- Design analysis methodology
- Benchmark analysis and criticality summary
- Power profiles
- Reactivity and control system
- Neutron and gamma–ray fluence
- Inherent safety features

PRATT & WHITNEY XNR2000 CERMET NTRE 25492

MODELS DEVELOPED TO ACCURATELY PREDICT REACTOR NEUTRONICS





PRATT & WHITNEY XNR2000 CERMET NTRE

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<u>Models Developed To Accurately Predict Reactor Neutronics</u>

A three dimensional model of XNR2000 core is developed. Thirty radial and azimuthal regions and 6 axial zones are used to model the details of the inner core, the outer core, the interfacial core reflector and the lateral support structure are included in the model. Six axial zones are used to address the axial temperature gradient in the inner and the outer cores. Group average neutron cross-sections for all 180 regions are generated at their average operating temperatures. Each region is divided to tens of finite volumes for the calculation of flux and effective multiplication factor.

DESIGN ANALYSIS METHODOLOGY TAILORED TO FAST SPECTRUM

- Multigroup cross–sections generated by COMBINE (ENDFB–V)
- MCNP (4.2) used for complex geometries
- BOLD VENTURE (3–D diffusion) used for power profile and reactivity
- ANISN (I–D ,Sn) used for analysis of hetrogeneous boundaries
- Results benchmarked with GE 710 testing
- Results independently confirmed by B&W and ANL

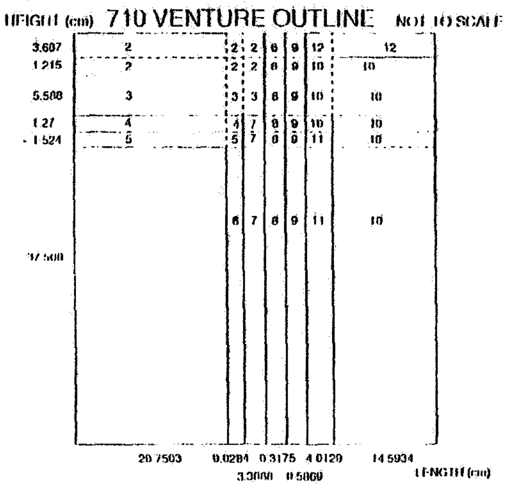
PRATT & WHITNEY XNR2000 CERMET NTRE　　25494

<u>Design Analysis Methodology Tailored To Fast Spectrum</u>

The major neutronic design analysis tools used for core physics studies include:

> COMBINE: A multigroup neutron cross-section generation code which combines the PHROG fast neutron library with the INCITE thermal neutron library (available through RSIC).
>
> BOLD VENTURE: A 3-D neutron diffusion code (available through RSIC).
>
> ANISN: A 1-D transport (Sn) code (available through RSIC).
>
> MCNP: A Monte Carlo code for stochastic analysis of the core criticality, power distribution and neutron and gamma-ray doses (available through LANL and RSIC).

VENTURE MODEL GENERATED FOR GE 710 MOCKUP 1A BENCHMARK



Material list

1 Core U, W, Ta, Al, O
2 Tube Sheet 303 SS
3 Tube Sheet and Mo 303 SS, Mo
4 Mo Transition Mo
5 Mo Plug Mo, Ta, W
6 Cladding Ta, W
7 Be (.85), Al
8 Shell 303 SS
9 Transition Al
10 Inner Reflector Zone Be (.85), Al
11 Outer Reflector Zone Be (.90), Al
12 Gap

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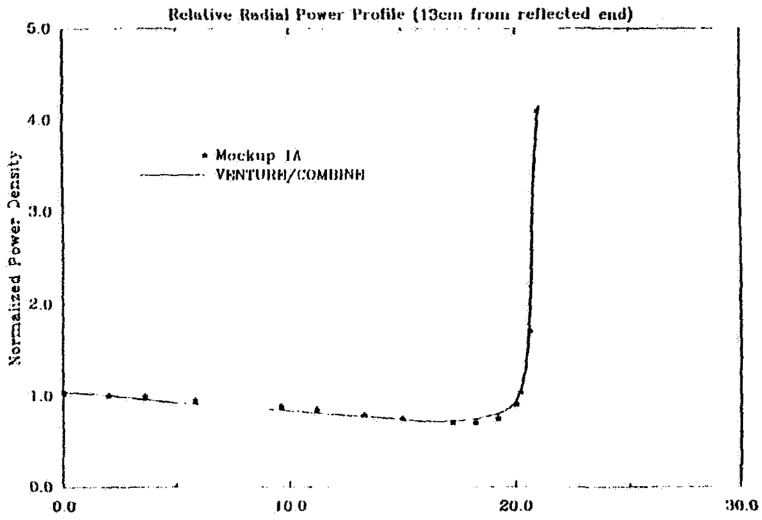
25495

<u>Venture Model Generated For GE710 Mockup 1A Benchwork</u>

GE710 program Mockup 1A critical configuration was used to benchwork the 3-D, 16-group COMBINE/VENTURE and continuous energy MCNP 4.2 models. Mockup 1A features core physics characteristics comparable with 25000 lbf XNR2000 engine design. The materials concentration and core dimensions are taken directly from the GEMP 442 report.

OUTSTANDING PREDICTION ACCURACY IS HARD TO BELIEVE



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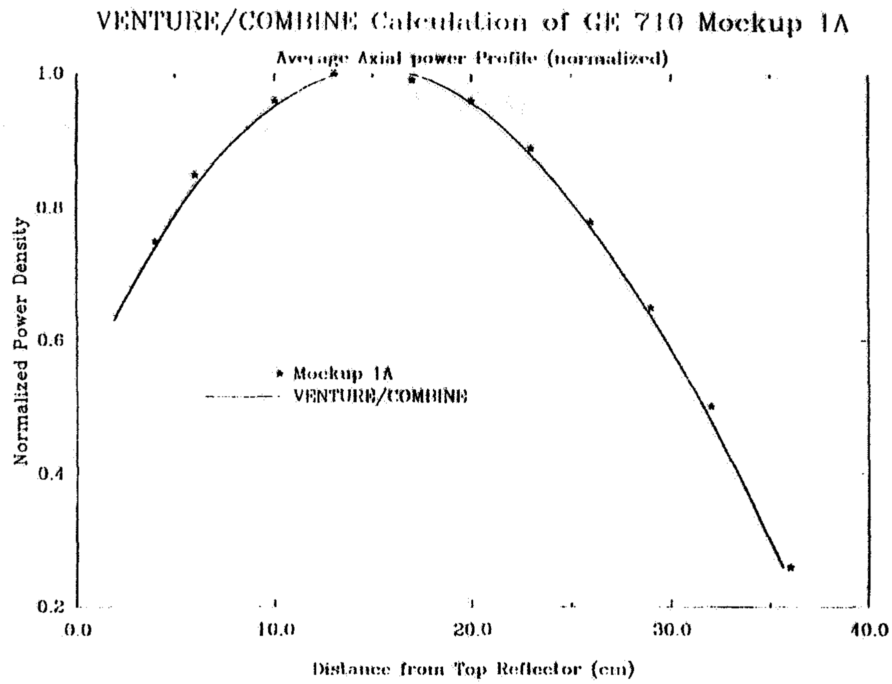
Outstanding Prediction Accuracy

VENTURE/COMBINE calculated values of the normalized radial power profile compares well with the GE710 experimental results, Both experimental and calculated power profiles are normalized to the power level at the radial distance of 2 cm from the core centerline. The calculated values of the radial power density beyond the last measurement point are not shown. The measured value of the relative power density is 4.1 where the COMBINE/VENTURE calculated maximum radial power density is 8.3. The maximum power density close to the reflector is very sensitive to the position.



VENTURE/COMBINE Calculation of GE710 Mockup 1A

COMBINE/VENTURE calculated values of the average axial power profile compares well with GE710 Mockup 1A experimental results. Two experimental points at the top and bottom of the reactor are excluded. Large uncertainty in experimental data at the unreflected end of the Mockup 1A reactor. Additionally, the VENTURE/COMBINE calculated value of K_{eff}, 0.991, compares well with the measured value of 1.000.

XNR2000 BASELINE CORE CRITICALLITY INDEPENDANTLY CONFIRMED

	Venture/Combine (P&W)	MCNP (P&W)	MCNP (B&W)	MCNP (ANL)
Keff	1.0183 (24 groups)	1.021	1.025	1.007
	1.0183 (16 groups)			
	1.0210 (12 groups)			
	1.0601 (8 groups)			
	1.0559 (4 groups)			

- Good agreement between 2–D, 16 groups diffusion calculation and MCNP
- Good agreement between independently performed MCNP calculations

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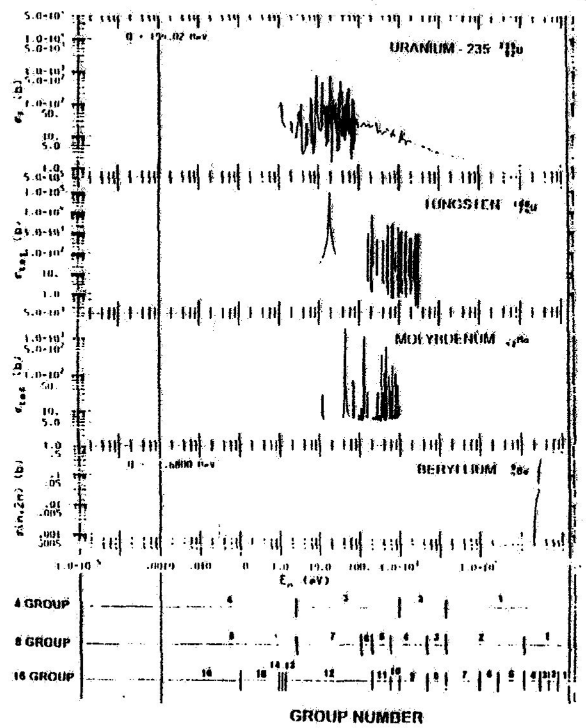
<u>XNR2000 Baseline Core Criticality Independently Confirmed</u>

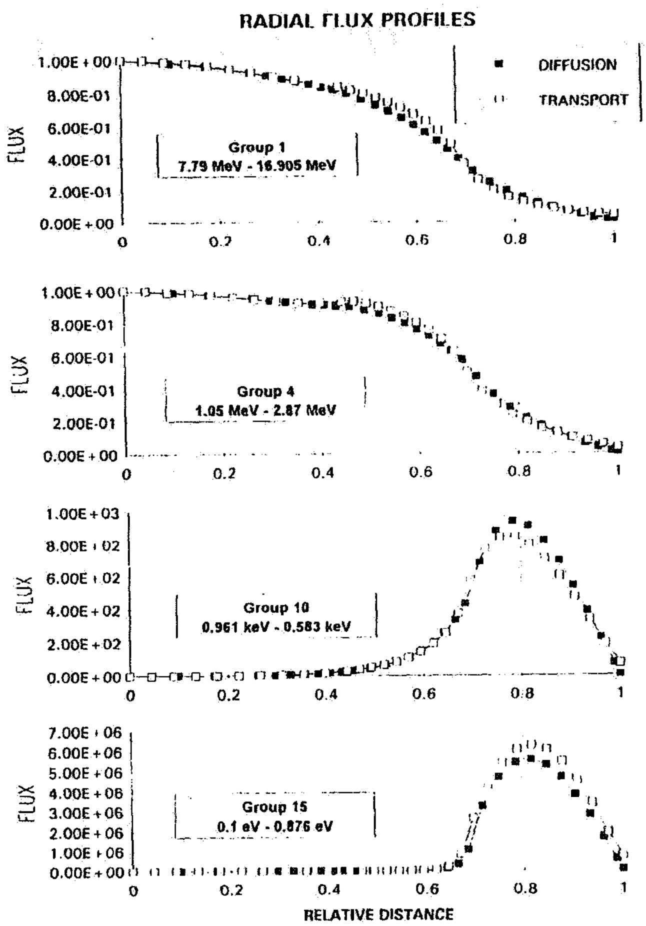
The 16 group COMBINE/VENTURE K_{eff} calculation of the XNR2000 core shows good agreement with MCNP calculated values of K_{eff}. Pratt & Whitney MCNP calculations are for a minimum of 500,000 histories. Babcock and Wilcox and Argonne National Laboratory calculations of XNR2000 core K_{eff} are based on a minimum of 100,000 histories. The small differences between MCNP calculated results are due to slightly different number densities and cross-section libraries used.

16 GROUP ACCURATELY MODELS SPECTRUM



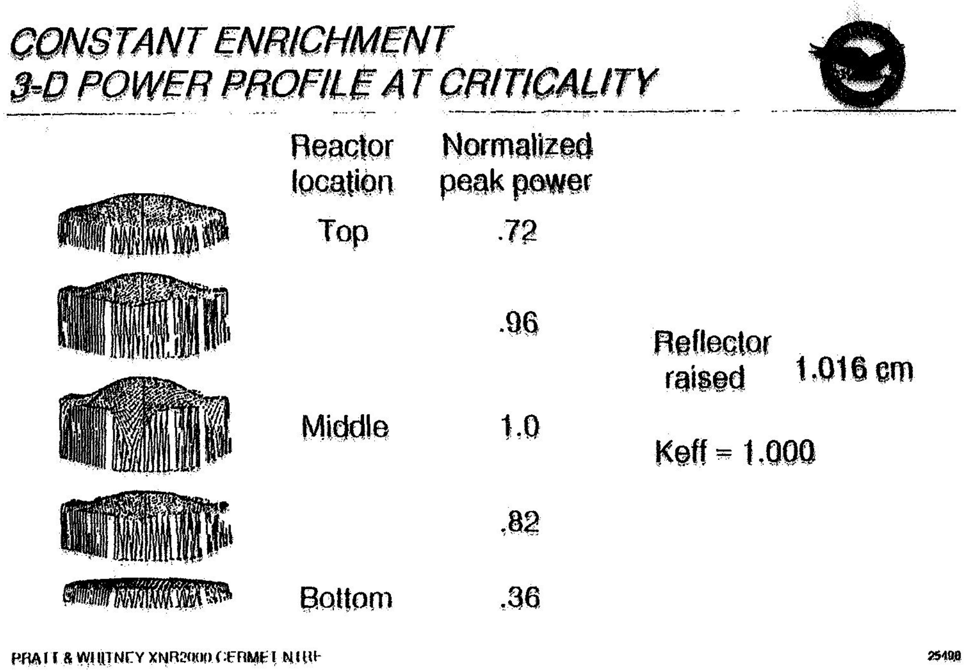
<u>16 Group Accurately Models Spectrum</u>

The selection of neutron energy groups is influenced by the location of isolated and non-isolated resonances of uranium, tungsten, molybdenum, and the energy threshold for the Be (n,2n) reaction. With a carefully selected energy partition, 12 group calculation proved to be adequate. The optimum choice of energy partition for 16 group calculation is shown and was used in all reactor studies.



<u>XNR2000 Radial Flux Profiles</u>

Radial flux profiles for the XNR2000 baseline core as presented. Both transport and diffusion theory are used to calculate total neutron flux for energy groups 1, 4, 10 and 16. The difference in the calculated value of flux at the vicinity of the reflector-core boundary is due to the fix law approximation used in the diffusion theory. There is also a noticeable difference between predicted values of flux for intermediate and low energy neutron groups in the reflector.

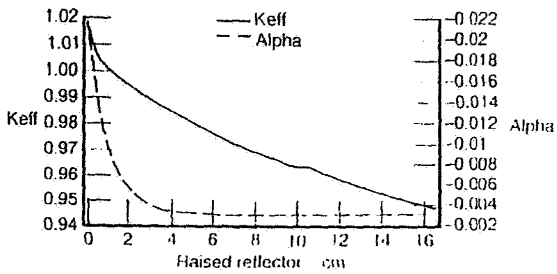


<u>Constant Enrichment 3-D Power Profile At Criticality</u>

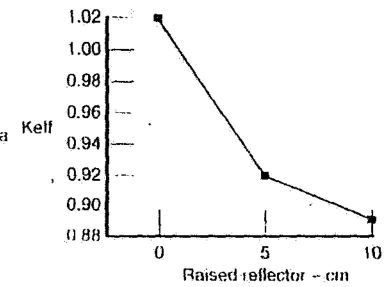
Radial power profile at different axial location of XNR2000 core. Ten degree reflector segments are raised for 1.016 cm to achieve K_{eff} = 1.0. Circumferential power tilt at the axial location of 50 cm is due to the raised reflectors.

HIGH REFLECTOR WORTH ENABLES ROBUST BASELINE CONTROL APPROACH

Reflector worth calculations



Shutdown calculations



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25501

<u>High Reflector Worth Enables Robust Baseline Control Approach</u>

The 25000 lbf XNR2000 baseline engine is powered by a compact fast reactor. The neutron leakage from the core region to the reflector is very significant. One of the options to control the reactor is the axial movement of the 10" reflector segments. Large reflector segments (50") can be used for large insertion of negative reactivity and reactor shutdown. The reflector worth calculations were conducted as a function of distance raised for a bank of six 10" reflector segments. The shutdown calculations were conducted for six 50" reflector segments.

REACTOR DESIGN PROVIDES ROBUST REACTIVITY AND CONTROL MARGIN

Reactivity effect	Reactivity% $\frac{\Delta k}{k}$
Temperature effect (30% 3000k)	$-.6 \pm .3$
Fuel burnup (6000 mw–hr)	$-.1 \pm .03$
Required excess Reactivity (maximum)	$+1.0$
Design excess reactivity	2.0 ± 0.5

Control system requirements	
Installed reactivity (maximum)	2.5
Minimum scram requirements	2.5
Minimum required control system worth	5.0
Design control system worth	10.0

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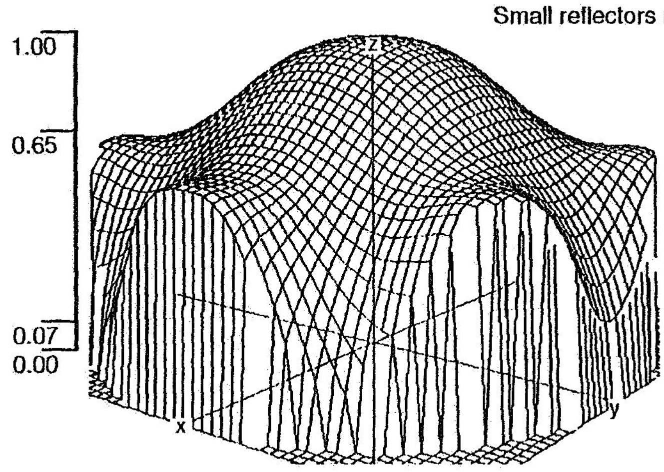
25500

<u>Reactor Design Provides Robust Reactivity and Control Margin</u>

COMBINE BOLD VENTURE computer code system is used to calculate the reactivity effect due to operational temperature and fuel burnup. High temperature cross-sections are generated and used to estimate the reactivity temperature effect at full power operational temperature. A total of 12 hours of full power operation is assumed to calculate the fuel burnup reactivity worth at the end of core life. The reactivity installed in the core is in excess of 2% which is needed to compensate for the loss of reactivity as the operation proceeds. The design control system worth is about -10%.

CONSTANT ENRICHMENT 3-D POWER PROFILE WITH RAISED REFLECTORS

Small reflectors raised 5 cm



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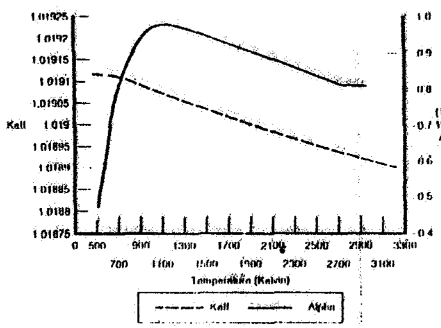
25502

__Constant Enrichment 3-D Power Profile With Revised Reflector__

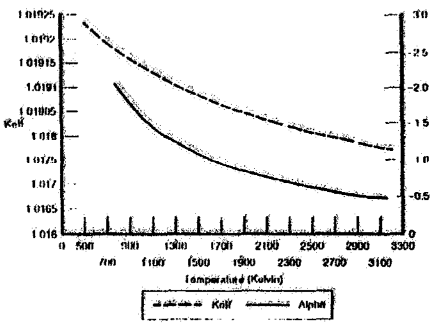
Power tilt due to the axial displacement of ten degree reflector segments is shown. The 10 degree reflector segments are raised by 5cm. The COMBINE BOLD VENTURE computer code system is used to calculate 3-D power distribution in the mid-core region. The power peaking at the core central axis is increased due to the significant leakage loss of neutrons through the opening in the radial reflector.

REACTOR HAS DESIRABLE NEGATIVE TEMPERATURE COEFFICIENT

Inner Core



Outer Core



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Reflector Has Desirable Negative Temperature Coefficient

- Temperature coefficient of reactivity is very small.
- Inner core fuel temperature coefficient is one order of magnitude smaller than the outer core fuel temperature coefficient.
- Outer core fuel temperature is comparable with GE710 Mockup 1A fuel temperature coefficient.

XNR2000

Inner Core: α @ 2200K) = -8.0×10^{-8} $\Delta K/K$ / °K

Outer Core: α @ 2200K) = -6.8×10^{-7} $\Delta K/K$ / °K

ADEQUATE INTERNAL SHIELDING INCLUDED IN DESIGN

Neutrons	XNR2000	NASA limits
Fast neutron flux (E>1.0 mev)	$(8.0 \pm 2.0) \times 10^{10}$	2.0×10^{12}
Intermediate energy neutron flux	$(2.4 \pm .6) \times 10^{12}$	3.0×10^{12}
Thermal neutron flux	$(3.6 \pm .9) \times 10^{11}$	6.0×10^{11}

Gamma – rays

Model results indicate gamma–ray fluence is very sensitive to system geometry. A refined estimation of gamma – ray fluence will require further definition of configuration and constraints

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Adequate Internal Shielding Included In Design

MCNP is due to calculate the fast, intermediate and thermal neutron fluxes at the upper part of the core. Fast and thermal neutron fluxes are significantly lower than the limits specified for the baseline design. Accurate estimation of the gamma-ray flux at the upper part of the shielded core require more detailed information on the upper core structural materials.

SUMMARY OF XNR2000 REACTOR NUCLEAR DESIGN

- State of the art analysis techniques employed to ensure design criticality, controllability and safety

- High confidence provided by benchmark analysis and independent evaluations by B&W and ANL

- Evaluation of all major reactor issues confirm advantages and flexibility of baseline approach

PRATT & WHITNEY XNR2000 CERMET NTRE

25507

N93-26920

Babcock & Wilcox Assessment of the Pratt & Whitney XNR2000

Babcock & Wilcox Space Systems Engineering

K.O. Westerman, S.W. Scoles, R.R. Jensen, J.R. Rodes, M.W. Ales

October 1992

This Document Prepared Exclusively for Pratt & Whitney

Babcock & Wilcox Assessment of the Pratt & Whitney XNR2000

Babcock & Wilcox Space Systems Engineering

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Babcock & Wilcox Assessment of the Pratt & Whitney XNR2000

Pratt & Whitney contracted with Babcock & Wilcox Advanced Systems Engineering/Space Systems Engineering to provide engineering support services for their NASA SEI Task Order Contract. Among other things, B&W is a reactor system vendor with physics, thermal hydraulics, materials, systems, mechanical engineering and manufacturing capabilities. B&W is also the operator of the only commercial facility licensed to manufacture large quantities of highly enriched reactor fuel.

Introduction

- **Scope of B&W Efforts**
  - **Fuel Element Fabricability Assessment**
  - **Mechanical Design Review**
  - **Neutronics Analysis Review**
  - **Safety Assessment**
- **Results of Mechanical and Physics Reviews Included in P&W and U of F Presentations**

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Introduction

B&W performed four subtasks for P&W as follows:

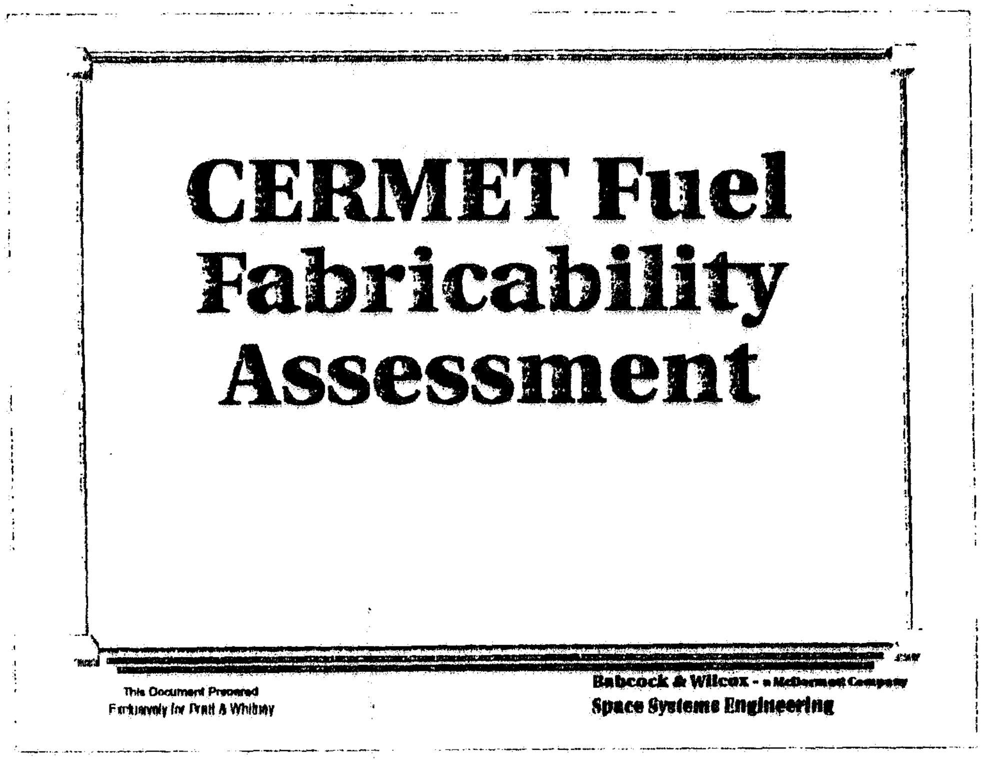
1) Fuel Element Fabricability Assessment - An assessment of the fabricability and manufacturability of CERMET fuel elements and the recoverability of the applicable technology.

2) Mechanical Design Review - An overall review of the reactor system from a mechanical engineering standpoint.

3) Neutronics Analysis Review - A review of the neutronics calculations performed for P&W by the University of Florida.

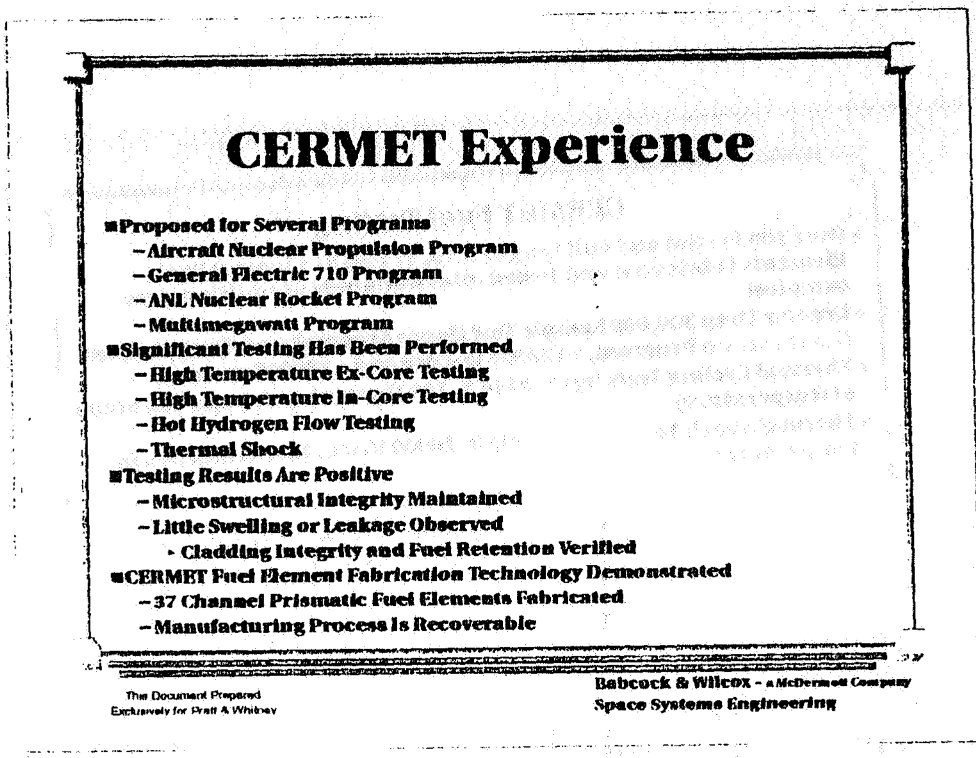
4) Safety Assessment - An overall assessment of the reactor system from a safety point of view.

The results of the mechanical and physics reviews have been integrated into the design and previously presented. The results of the fuel and safety assessments are presented here.



CERMET Fuel Fabricability Assessment

The fabricability of CERMET fuel elements is a major issue in the design of the XNR2000 reactor. The reactor uses both tungsten and molybdenum based UO_2 CERMET fuel elements. Most work on CERMET fabricability has focussed on tungsten based fuel elements. Since tungsten based CERMET fuel elements are more difficult to fabricate, it is reasonable to assume that if they are fabricable, then molybdenum based CERMET fuel elements will also be fabricable. The same issues and considerations that apply to tungsten based fuel elements will also apply to molybdenum based fuel elements.



CERMET Experience

CERMET based fuel elements have been proposed for several programs in the past. The Aircraft Nuclear Propulsion program provided early experience with CERMETs. Some fuel was made and tested, but it was not a prismatic form. In the General Electric 710 program, prismatic 37 coolant channel CERMET fuel elements, similar to those proposed for use in the XNR2000, were constructed using at least two different techniques. Extensive testing was performed and documentation of these efforts is good. The Multimegawatt program demonstrated recovery of the ANL Nuclear Rocket Program technology. During the course of the 710 program and the Nuclear Rocket Program, significant testing of CERMET fuels was performed including: high temperature ex-core testing, high temperature in-core testing, hot hydrogen flow testing and thermal shock testing. The test results were positive. Little swelling or leakage was observed and microstructural integrity was maintained. CERMET fuel cladding integrity and its ability to retain the fuel was also verified.

The CERMET technology development that has been performed forms a good basis for the necessary follow-on work. The past work should be integrated with current technology, where appropriate, and a demonstration fuel element should be fabricated using depleted uranium or a surrogate fuel material.

CERMET Fuel Testing

- **Over 100 Partial and Full Length 7, 19, 37 and 91 Channel Fuel Elements Fabricated and Tested (plus hundreds of additional test samples)**
- **Greater Than 300,000 Sample Test Hours Accumulated (Fuel Element Qualification Program, >120000 in-core and >180000 ex-core)**
- **Thermal Cycling Tests (up to 2444 K, 100 thermal cycles and 100 hours at temperature)**
- **Thermal Shock Tests (in-core, up to 16000 K/sec, 2870 K maximum temperature)**

Selected Test Results

	In-Core (LTFF 9)	In-Core (LTFF-11)	In Core (4 samples) (E1R2-6)	Ex-Core Hydrogen Flow	Ex-Core Hydrogen Flow	Ex-Core Hydrogen Flow
Temperature (K)	2273	1923	1873 to 2298	2773	2973	3073
Time (hours)	1015	3086	to 537	1000	50	10
Power Density (MW/l)	15	20	1 0 to 2 1	-	-	-
Burnup (atom %)	49	90	0 5 to 1 1	-	-	-
Results	Negl Swelling	Negl Swelling	Some Leak & Swelling	No Failures	No Failures	No Failures

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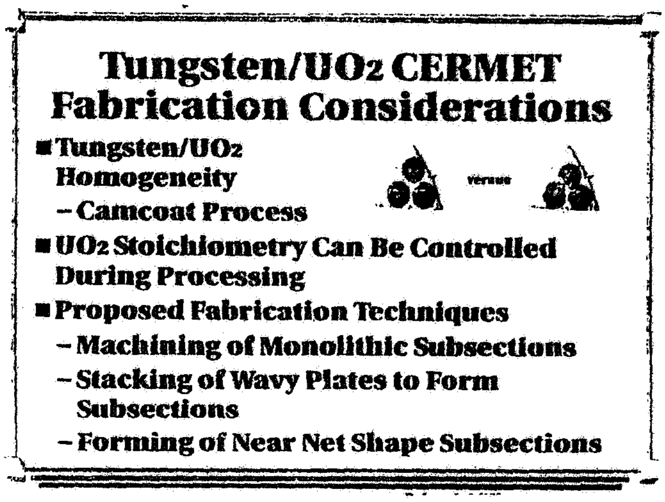
Basis of B&W Fuel Assessment

- Program Reports
  - General Electric 710 Program
  - ANL Nuclear Rocket Program
  - Multimegawatt Program
- Visit to Argonne National Laboratory
- Discussions with B&W Fuel Manufacturing Experts
  - Experienced With Similar Materials and Processes (refractory metals and UO_2)

Basis of B&W Fuel Assessment

The B&W fuel fabricability assessment is based on written accounts of previous work, discussions with people who performed some of that work and discussions with our own manufacturing experts. The General Electric 710 program and the Multimegawatt program both left good documentation of their efforts. A trip to Argonne National Laboratory was made to talk to some of the people involved in the manufacture of CERMET fuel. B&W manufacturing personnel are experienced with refractory metals and UO_2. Discussions with B&W fuel manufacturing experts solidified confidence that CERMET fuel manufacturing technology is easily recoverable.



Tungsten/UO_2 CERMET Fabrication Considerations

The CERMETs used for the XNR2000 are formed by consolidation and densification of UO_2 and tungsten (or molybdenum) powders. The UO_2 in CERMET fuel elements produced using any process must be distributed uniformly in the element. Uniformity as used here has two different meanings. First, the UO_2 loading must be locally uniform throughout the element subsection. Second, and perhaps more important, the UO_2 fuel particles must not cluster in the CERMET, but rather must be individually isolated by tungsten matrix material. This ensures that each fuel particle will be cooled adequately. Because of the differences between the behavior of tungsten and UO_2 powders, these powders must be pre-processed to ensure blending before they can be consolidated in a powder based process. It is assumed in the discussion of each consolidation process that a suitable blending process has been used prior to the actual fabrication of the element. One possible blending process is the Camcoat process developed in the General Electric 710 program.

A number of possible consolidation processes may be employed. These include, but are not limited to, pressing and sintering, hot extrusion, high energy rate forming (HERF) and heat treating, hot pressing and hot isostatic pressing (HIP). All of these processes should be capable of providing solid element subsections with little porosity in the matrix. Some of the processes, such as hot extrusion, will impart an axial texture to the element material. The acceptability of such texture must be evaluated prior to selection of a texture producing process.

In all of the above consolidation processes, the stoichiometry of the UO_2 fuel can be controlled. This is done by performing the consolidation operation in an atmosphere where the oxygen partial pressure is controlled. Typically, this involves consolidating in a hydrogen based atmosphere. Control of UO_2 stoichiometry is critical because deleterious effects occur if the fuel is either hyper- or hypo- stoichiometric.

Three CERMET fuel element subsection fabrication techniques were evaluated: machining of monolithic subsections, stacking of wavy plates to form subsections and forming of near net shape subsections. Each technique is described and a preferred fabrication technique is recommended. Machining of monolithic subsections was used in the General Electric 710 program and forming of near net shape subsections was used in the ANL Nuclear Rocket Program to fabricate fuel elements.

Fuel Element Assembly

- **Components**
  - CERMET Fuel Subsections
  - Tungsten/Rhenium Structural Can
  - Tungsten Coolant Channels
- **Assembly Options**
  - Stack Subsections In Can, Insert Tubes For Coolant Channels, HIP To Bond Can and Tubes To Fuel
  - Diffusion Bond Subsections To Each Other, HIP To Bond Fuel To Can, Form Coolant Channels By CVD Coating or Insertion of Tubes

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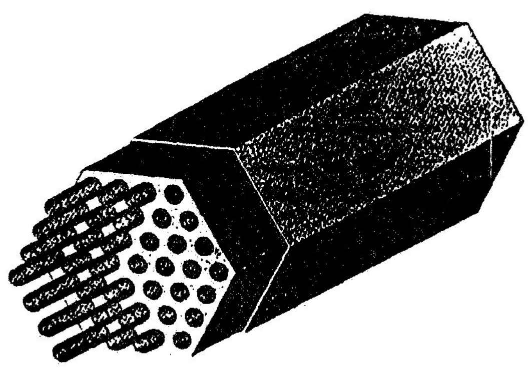
Fuel Element Assembly



Fuel Element Assembly

A complete CERMET fuel element consists of the CERMET fuel subsections, a tungsten/rhenium structural can and tungsten lined coolant channels. Final assembly of these components into full length fuel elements can be accomplished in a number of different ways. One option, used in the 710 Program, is to stack the subsections, insert tungsten flow tubes in the aligned channel holes and place the stack in the tungsten/rhenium can. A HIP operation is performed to bond the can and flow tubes to the subsection stack. With this option, the can is structural and is the sole load bearing component in the element. Another option is to coat the external surfaces of the element subsection and the ID of the coolant channels with tungsten prior to bonding a structural can onto the stack. This would eliminate the need for inserting full length flow tubes into the elements. A final option would be to bond the subsections together to form an integral fuel stack prior to bonding the stack into the can. Diffusion bonding is one possibility for the bonding process. Tungsten washers or standoffs could be used between subsections to create a plenum or transition section to minimize the effects of hole misalignments. The coolant channels can be formed as coatings on the subsections prior to assembly or by inserting tubes prior to can bonding.

Of the assembly options, bonding the subsections together prior to further assembly is preferred because it reduces the dependence on the fuel element can for structural integrity. It is not clear what technique is best for forming the tungsten flow tubes. Further technology evaluation is necessary in this area.

Monolithic Fabrication

- Consolidate and Densify a Monolithic Subsection
- Machine Coolant Channel Holes
  - Diamond Drilling or Ultrasonic Machining (EDM won't work)
  - Subsection Thickness Limited By Runout To 1 to 5 cm (1 cm sections with smaller holes were made in the GE 710 program)
- If Using CVD Coated Coolant Channels, Apply Coating Prior To Further Assembly

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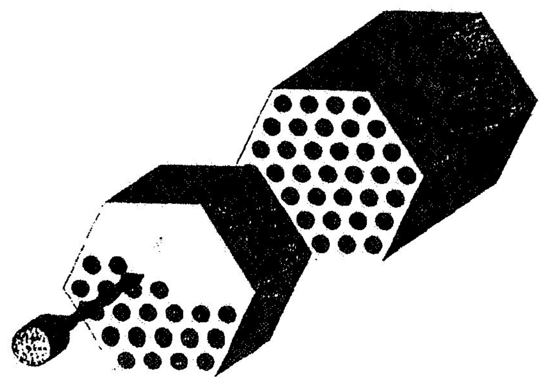
Monolithic Fabrication



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Monolithic Fabrication

In the monolithic process, the coolant flow channels are machined into a consolidated subsection. This may be done by drilling with diamond tooling or ultrasonic machining. Electrical discharge machining (EDM) is not feasible because of the high volume fraction of insulating oxide fuel in the CERMET. The maximum element subsection length which can be processed is limited by runout and depends on hole size, fuel loading, hole pitch and manufacturing tolerances. The maximum subsection length must be determined as part of the technology development. Experience with other materials suggests that this length is between 1 and 5 cm. This process was used in the General Electric 710 program to produce 1 cm thick CERMET subsections with smaller holes than proposed for the XNR2000.

Monolithic Discussion

- **Advantages**
  - **Machining of Coolant Holes Can Start From True Positions for Each Subsection**
  - **Short Subsections Simplify Inspection**
- **Disadvantages**
  - **Large Amount of Waste Generated By Machining**
  - **Machining Exposes UO_2**
    - **Fuel Loss**
    - **Coating Difficulties**
  - **Limited To Short Subsections By Runout**
    - **Joining of Many Subsections Challenging**

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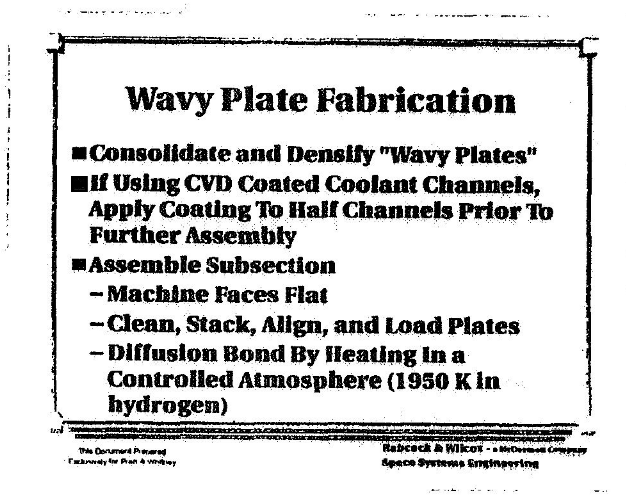
Monolithic Discussion

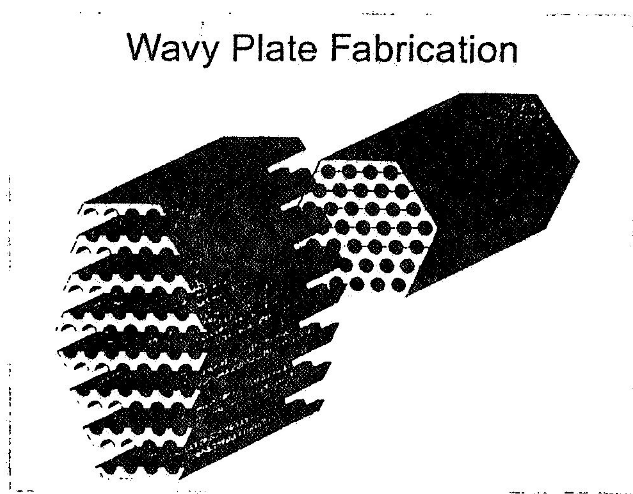
The major advantage of the Monolithic process is that the coolant channels can be machined into the element subsections starting from true positions. Runout will increase as the depth of the hole increases, ultimately limiting the length of the element subsection which can be processed. The short length of the subsections is an advantage for inspection if the subsection flow tubes are applied as coatings before element assembly, because it will be easier to verify the integrity of the flow tube.

There are a number of disadvantages associated with the Monolithic process. As previously mentioned, subsection length is limited to between 1 and 5 cm. Stacking and bonding of the many subsections necessary to make a complete fuel element would be complicated.

Another disadvantage of the Monolithic process is that a large amount of scrap tungsten/UO_2 debris will be generated by the machining process. The uranium must be recovered from this debris. In the current design, 45% of the UO_2 initially in the consolidated fuel element subsection ends up as debris.

A final disadvantage of the Monolithic process is that the machining process exposes UO_2 fuel particles on the channel surface. This is an important effect for the following reasons. First, a significant fraction of the total amount of fuel in an element subsection will be exposed in the coolant channels. Assuming an average UO_2 particle size of 100 μm, the fraction of fuel within one-half of a particle diameter of a coolant channel is 4.9% for the current design. It is not unreasonable to assume that a large portion of the exposed fuel particles would be damaged in the channel machining process and be lost in debris, especially if fuel particles are intentionally porous (to collect gaseous fission products). This would lead to a relatively rough coolant channel surface and possibly to an unacceptable loss of fuel. A second consequence of having exposed fuel on the channel surfaces is that it may be difficult to form the coolant flow tubes by CVD. Porous or damaged fuel particles may trap halide feed material or CVD byproducts and compromise flow tube adhesion during operation. In addition, UO_2 may react with the CVD feed material or byproducts to an unacceptable degree.





Wavy Plate Fabrication

A fabrication technique proposed but not actually used in the General Electric 710 program was to build element subsections from flat plates. This fabrication process begins with the formation of tungsten/UO_2 powder compacts in the form of plates. Each plate would be fabricated with grooves on both of its faces. The grooves would be semicircular and correspond to one-half of a coolant channel. Stacking and aligning the plates would form an element subsection with complete, circular channels. This technique can produce elements with coolant channels arranged on a square or triangular pitch, by varying the offset between the groove patterns on the opposite faces of the plates.

After the formation of the powder compact, the plate is then consolidated by sintering. It may also be possible to perform the consolidation by hot pressing, if a suitable material for the fixturing required can be identified. Following consolidation, the plate would probably need to be ground to ensure flatness. The plate could then be coated with tungsten to form a coating on the half channels. When the plates are assembled, this coating would form the flow tube. Alternatively, the tube walls could be formed after subsection or fuel element assembly by CVD coating or insertion of tubes and HIP. After being stacked, aligned and loaded, the subsection is diffusion bonded by heating in a controlled atmosphere (1950°K in a hydrogen atmosphere for 1 hour for example).

Wavy Plate Discussion

- **Advantages**
  - **May Not Require Machining Of Coolant Channels**
  - **Inspection Simplified By Thin Sections and Exposed Coolant Holes**
- **Disadvantages**
  - **Close Tolerances May Be Difficult to Maintain (channel position +/- .02 mm, stacking alignment +/- .02 mm)**
  - **Minimum Amount of Machining Required Prior to Joining**

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Wavy Plate Discussion

One of the major advantages of the Wavy Plate process is that the coolant channels can be fabricated without having to machine them into fully consolidated CERMET. Another major advantage is that forming the tungsten flow tubes by coating the plates prior to stacking would allow easy and detailed inspection of the integrity of the flow tube.

The major disadvantage of this process is that sintering induced shrinkage will affect the final dimensions of the plate. Accordingly, it may be difficult to maintain the required tolerances in plate dimensions. Additional difficulties may be encountered in stacking and aligning plates to form element subsections. Even if plate dimensions are such that perfect alignment is achieved, it may be difficult to maintain this alignment during the plate bonding operation. Channel position and stacking alignment tolerances will both be of the order of ± .02 mm. The tight tolerances are necessary to minimize coolant channel offsets and maximize web contact area. A final potential problem is that, if additional machining is required after consolidation, many of the disadvantages related to machining mentioned earlier may be present.

Near Net Shape Fabrication

- **Make a Mold and Insert Molybdenum Rods Where Coolant Channels Will Be Located**
  - **Maximum Thickness Limited By Molybdenum Rod Stiffness and Straightness (40 to 50 cm sections made in Multimegawatt program)**
  - **Ultimately, Thickness Limited To 5 to 15 cm By Other Considerations**
- **Fill Mold With Tungsten/UO_2 Powder**
- **Cold Isostatic Press**
- **When Pressure Is Released, Spring-Back Provides Enough Clearance (several mils) To Allow Removal Of Molybdenum Rods**
- **Sinter At 1950 K**
- **If Using CVD Coated Coolant Channels, Coat Prior To Further Assembly**

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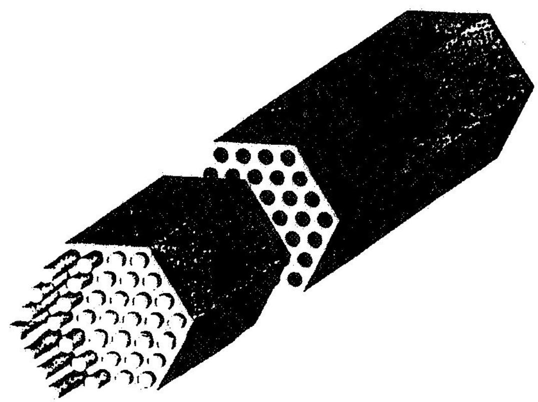
Near Net Shape Fabrication



Near Net Shape Fabrication

Fuel elements were constructed using this process in the General Electric 710 program and again during the Multimegawatt program. In this process, treated tungsten/UO_2 powder is introduced into a flexible rubber mold which contains molybdenum rods or wires. After filling, the mold is cold isostatically pressed to form a fuel element subsection powder compact. The molybdenum rods or wires are used to form the coolant channels in the compact. During mold filling, the molybdenum channel formers are held rigidly in a triangular array, with the pitch between the channel formers slightly greater than that required in the consolidated subsection, to allow for shrinkage during sintering. The key to this process is that elastic strain stored in the powder compact during isostatic pressing causes the compact to expand slightly after the pressure is removed. This spring-back effect is of a magnitude sufficient to allow the channel formers to be removed easily from the compact.

After isostatic pressing, the powder compact is sintered. The tungsten matrix can be densified to essentially theoretical density at the relatively modest temperature of 1950°K. CERMETs containing up to 61 volume percent UO_2 have been fabricated.

Segments 40 to 50 cm long were made using this process during the General Electric 710 program. The useful length may ultimately be limited to 5 to 15 cm by other factors such as channel straightness tolerances and inspectability.

Near Net Shape Discussion

- **Advantages**
  - No Machining Required
  - Possible To Fabricate Longer Subsections
- **Disadvantages**
  - Sintering Shrinkage Must Be Considered
  - Inspection of Longer Subsections More Difficult

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<u>Near Net Shape Discussion</u>

This process has two major advantages. First, a fuel element subsection is produced which is truly near net shape. Some machining of the external surfaces of the subsection may be required if inter-element spacing tolerances are tight. However, the coolant channels are formed without machining and no fuel particles are exposed in the channels. The other advantage is that this process can fabricate significantly longer element subsections which drastically reduces the number of bonds required to fabricate an integral full length element.

The major disadvantage of this process is that significant process qualification and control will have be performed to ensure that dimensional tolerances in the consolidated element subsections will be met. This work is necessary to guarantee that sintering shrinkage is reproducible from run to run during production. Process optimization may be necessary for each batch of powder used. Another disadvantage of this process is that the longer length subsections produced make inspection of the coolant channel surfaces more difficult.

Inspection and Q/A Requirements

- **Define Fuel Element Specifications and Tolerances**
- **Validate the Chosen Process Using Destructive Testing To Verify and Quantify**
  - **Homogeneity**
  - **Uranium Assay**
- **Use Nondestructive Testing Techniques To Check For**
  - **Gross Defects**
  - **Dimensions (Size, Shape, Straightness, Roughness)**
  - **Bond Integrity**

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Inspection and Q/A Requirements

Significant development effort will be required in the areas of inspection techniques and quality assurance procedures. For critical items, such as fuel elements in a man-rated system, there is no such thing as an excessive amount of inspection. Manufacturing tolerances and specifications for fuel element quality must also be determined.

One obvious area for inspection is conformance to dimensional requirements. Aside from the more obvious dimensional measurements, measurements of coolant channel straightness and surface roughness must be performed. Both of these attributes would be expected to affect thermal hydraulic behavior.

Before element subsections are assembled into full length fuel elements, it will be necessary to verify that they are structurally sound. At a minimum, the porosity of the tungsten matrix should be determined and the absence of gross defects verified. Also required is measurement of the integrity of all bonds in the fully assembled fuel element. These include the bonds between element subsections, between the coolant flow tube and the CERMET fuel and between the can and the fuel element. Ultrasonic and eddy current inspection techniques can be used for these measurements.

Where direct measurements are not possible, verification has to be performed by qualifying the process. This is accomplished by running process control samples through the element fabrication process and performing destructive evaluations on them. Two measurements for which this may have to be done are UO_2 content and homogeneity of the fuel subsections. The necessity for homogeneity has already been discussed. Fuel content is required for SNM accountability. It is also necessary to verify the fuel loading of each element to ensure that the reactor will have sufficient reactivity.

Fuel Element Assessment Conclusions

- **Recommended Baseline Fabrication Approach - Forming of Near Net Shape Subsections**
  - **Demonstrated In Two Previous Programs**
  - **Well Documented Process**
  - **Technology Recoverability Demonstrated**
  - **Other Fabrication Approaches Still Viable As Backup Options**
- **No Materials Incompatibilities Noted**
- **Fuel Element Performance Limited By Melting Point Of UO_2**
- **Further Process Qualification and Development of Inspection Techniques Required**

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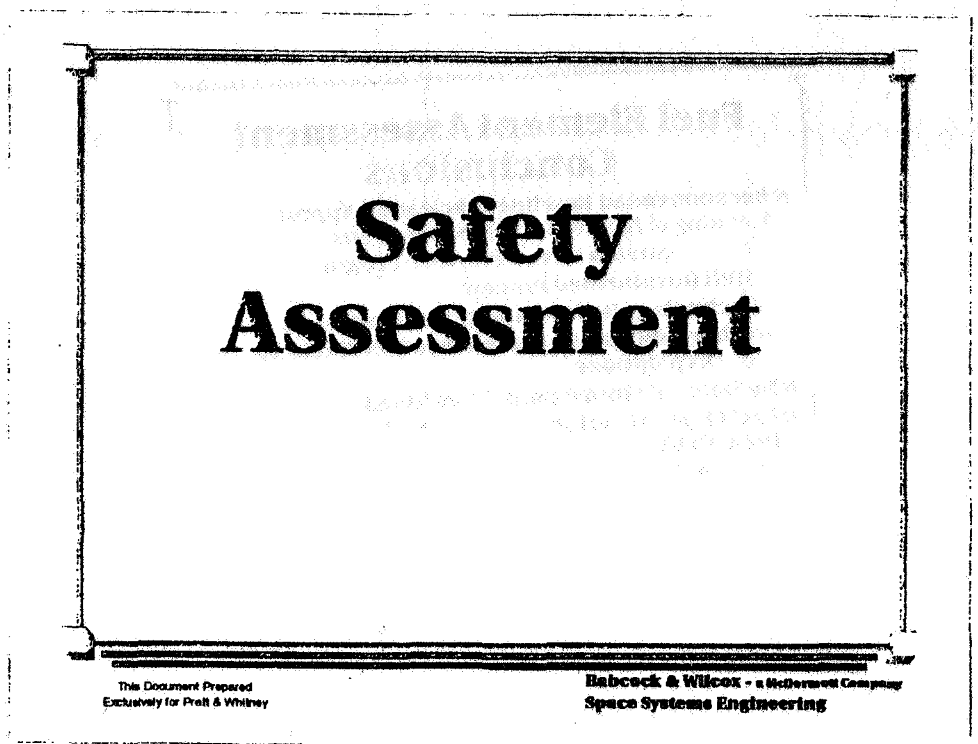
Fuel Element Assessment Conclusions

There are no insurmountable obstacles to fabrication of tungsten or molybdenum UO_2 CERMET fuel elements. XNR2000 fuel is manufacturable using demonstrated technology. There were no materials incompatibilities noted in this investigation. Fuel element performance is not limited by structural considerations but by the melting point of the UO_2. The maximum nominal fuel temperature in the XNR2000 is well below the UO_2 melting point.

The recoverability of the CERMET processing technology has been demonstrated. The development of CERMET fuel technology should not impose cost or schedule limitations.

Of the three processes considered for the fabrication of tungsten (or molybdenum) based UO_2 CERMET fuel elements, the Near Net Shape process is preferred. As discussed, this process has the potential of producing long length element subsections without having to machine the coolant channels. The process uses well known, technically simple processing steps. These steps will, of course, have to be extremely well characterized, to control sintering shrinkage and allow dimensional tolerances to be met. Finally, this process has been investigated extensively in the past and there is a significant experience base with it.

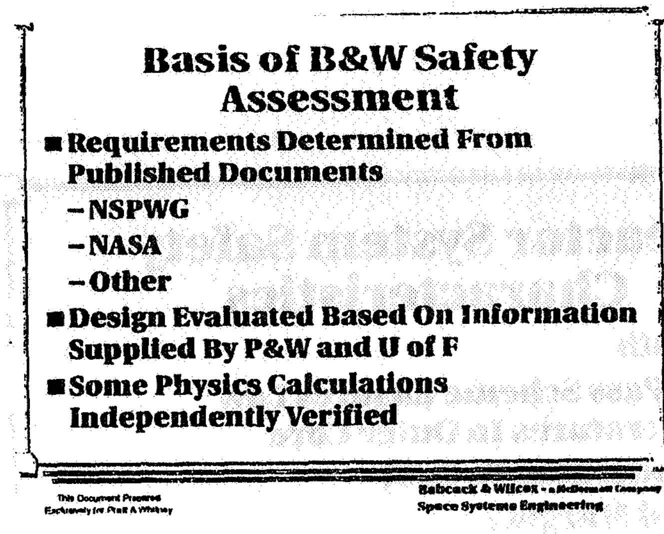
The wavy plate fabrication scheme also has potential. It should be further investigated in parallel as a backup option.



Safety Assessment

The Pratt & Whitney XNR2000 NTRE has been designed with safety as a primary consideration.

The safety of the public, mission personnel, the crew and the terrestrial and non-terrestrial environment have all been considered. The main safety characteristics of the XNR2000 are highlighted, and the effect of thrust level on safety is considered.



Basis of B&W's Safety Assessment

The first step in the B&W safety assessment was to determine what the requirements and safety concerns for SEI NTRE systems are. Some of the documents used are listed here:

A.C. Marshall, et al; "Nuclear Safety Policy Working Group Recommendations on Nuclear Propulsion Safety for SEI"; (to be published); 1991.

Nuclear Thermal Rocket Engine Requirements, Revision 3; NASA N.P. #002; 1992.

L.W. Connell, D.L. Potter, C.C. Wong and M.W. Kniskern; "Nuclear Thermal Rocket Entry Heating and Thermal Response Preliminary Analysis"; Proceedings of the Ninth Symposium on Space Nuclear Power Systems; p. 923; January 12-16, 1992.

Occupational Safety and Health Administration, Department of Labor; US Code of Federal Regulations, 29 CFR 1910.96.

S.N. Jahahan; "The Reactor Physics Design of Gas-Cooled CERMET Reactors and Their Potential Application to Space Power Systems"; Nuclear Technology; vol. 98, p. 257; 1992.

P.M. Sforza, M.L. Shooman, D.G. Pelaccio; "A Safety and Reliability Analysis for Space Nuclear Thermal Propulsion Systems"; IAA-92-0376; 43rd Congress of the International Astronautical Federation; 1992.

D. Atkinson, et al; "Collision Damage to Nuclear Satellites"; Proceedings of the American Nuclear Society Topical Meeting on Nuclear Technology for Space Exploration; p. 843; 1992.

D. Buden; "Safety Questions Relevant to Nuclear Thermal Propulsion"; Proceedings of the Ninth Symposium on Space Nuclear Power Systems; p. 648; 1992.

The XNR2000 design was then evaluated with respect to these requirements and concerns based on information supplied by Pratt & Whitney and the University of Florida.

Reactor System Safety Characteristics

- **Flow Path**
  - **Dual Pass Scheme Ensures Low Temperatures In Outer Core**
  - **No Moderator To Cool**
- **Thermal Margins**

Reactor Temperatures (K)

	Outlet Hydrogen	Peak Fuel	UO2 Melt	Refractory Melt
Outer Core	1659	2007	~3150	2900
Inner Core	2669	2880	~3150	3700

- **Control/Shutdown Redundancy**

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Reactor System Safety Characteristics

Safety considerations must be an integral part of the design process for any man-rated space system. Inherent safety is the preferred goal, and passive systems are preferred over active systems. The dual pass flow scheme ensures low temperatures in the outer core in a simple way. The fact that there is no moderator to cool also simplifies the flow path. The peak fuel temperature in the inner core is 2880°K (outlet hydrogen temperature of 2669°K); this compares to tungsten and UO_2 melting points of 3700°K and 3150°K respectively. In the outer core, the peak fuel temperature is 2007°K (outlet hydrogen temperature of 1659°K); the melting point of molybdenum is 2900°K. These large thermal margins provide advantages in transient and accident situations. The reflector windowing leakage control scheme proposed for the XNR2000 is simple and robust. Redundant control and shutdown systems are provided.

CERMET Fuel Form Safety Characteristics

- Thermal Shock Resistance
- Long Term Stability
  - Compatible With Hot Hydrogen
  - Low Swelling
- High Thermal Conductivity
  - Tungsten - ~100 W/m*K
  - UO_2 - ~2 W/m*K
  - Bulk CERMET - ~33 W/m*K
- Tungsten Used As Primary Barrier For Fission Product Retention
  - Demonstrated Performance In Hot Hydrogen
  - Low Tungsten Diffusion Coefficient
  - Low Volatilization To Vacuum

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CERMET Fuel Form Safety Characteristics

The XNR2000 design benefits from the inherent stability, robustness and transient tolerance of the CERMET fuel form. This may be a particular asset in the long dormant phase and in assuring operability until disposal. Tungsten and molybdenum CERMET fuels are more resistant to hydrogen erosion than carbide fuels, and therefore, lifetime is not limited by coatings technology. They also exhibit low swelling and are effective at retaining fission products. CERMET fuels exhibit excellent thermal shock resistance over a wide range of conditions. The high thermal conductivity of the fuel is advantageous in under-cooling scenarios and decay heat removal. Bulk tungsten/UO_2 CERMET thermal conductivity is in the range of 33 W/m*K, based on tungsten and UO_2 thermal conductivities of 100 and 2 W/m*K respectively. When damage thresholds are exceeded, the CERMET fuel form can handle significant degradation before failing catastrophically.

Tungsten is effective at retaining fission products. Its inherent stability, performance in hot hydrogen, low volatility to vacuum and low diffusion coefficient all combine to make it one of the best materials for this purpose. The General Electric 710 program demonstrated that the CERMET matrix alone will retain 85% of the fission products it contains. With intact tungsten cladding retention approaches 100%.

Fast Spectrum Safety Characteristics

- Negligible Hydrogen Worth (< .2 % ρ)
- Less Excess Reactivity Required
  - Lower Delayed Neutron Fraction Makes Given Reactivity Insertion Worth More
  - Negligible Xenon Reactivity Effects

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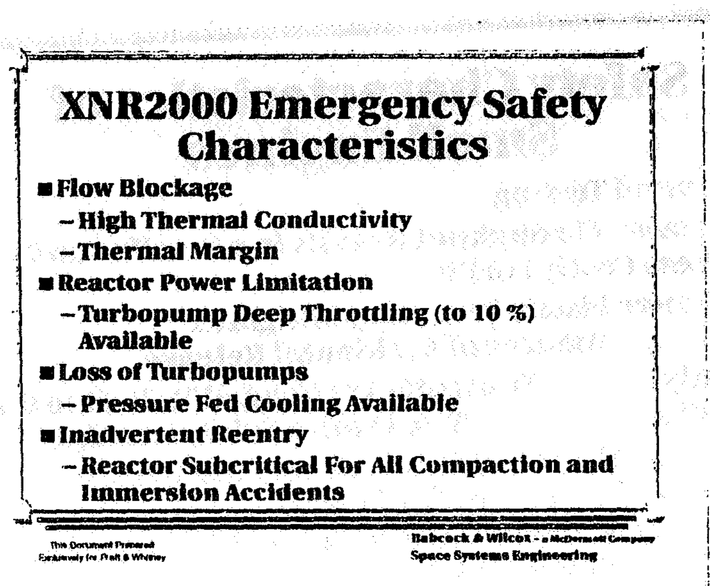
Fast Spectrum Safety Characteristics

The fast spectrum of the XNR2000 has several positive safety effects. The worth of the hydrogen in the XNR2000 is negligible (<.2 % ρ as calculated by the University of Florida and verified by Babcock & Wilcox). This is helpful at reactor startup, since there will be no large reactivity insertion due to the cold hydrogen. Little excess reactivity is required in the XNR2000. The lower delayed neutron fraction in the fast spectrum makes a given reactivity insertion worth more in terms of reactor response. Also, there a negligible xenon reactivity effect due to the fast spectrum, so there is no need to provide excess reactivity to overcome a large xenon transient.

The fast spectrum of the XNR2000 may also affect ground testing. Ground test facilities will have to be designed to handle the fast spectrum leakage from the NTRE.



<u>XNR2000 Emergency Safety Characteristics</u>

The high thermal conductivity and structural stability of the CERMET fuel is a benefit in accident situations. For partial or full flow blockage in a coolant channel, the high thermal conductivity mitigates the temperature rise. Temperatures around a blocked channel may approach or exceed the melting point of UO_2. This will not be a problem in the localized area involved. The tungsten can easily contain molten UO_2.

A loss of turbopumps accident is handled by pressure fed cooling. A high pressure reservoir will provide the hydrogen flow necessary for the critical period immediately following shutdown. After the first minute or so, the reactor can be cooled by feed tank pressure.

No NTRE currently under consideration can survive a full-power total loss of coolant accident. A low-power total loss of coolant would cause rapid shutdown of the reactor due to the negative reflector temperature coefficient. A total loss of coolant at very low power or during decay heat removal might not be catastrophic for the XNR2000 due to the robustness of the CERMET fuel.

The XNR2000 turbopumps can be throttled to 10% of full flow. This is a safety advantage for cases where the reactor power is limited for some reason. This feature allows the NTRE to provide reduced thrust at nearly full I_{sp}. This enables mission completion or full abort capability for a limited reactor power scenario.

It has been shown by the University of Florida, and verified by Babcock & Wilcox, that for all the accidents of concern in an inadvertent reentry (compaction and submersion) the reactor remains subcritical by a significant margin.

Safety Characteristics of Small Engines

- **Ground Testing**
  - Lower Throughput Results in a Smaller and Less Costly Facility
  - Lower Fissile Inventory Mitigates Consequences of Accidental Release
- **Early Design Tradeoffs Are Needed To Define the Optimum Engine Size From a Safety Standpoint**
  - Tradeoffs Should Include Mission Parameters (total thrust/length of burn)
  - Effect of Engine Size/Number of Engines on Redundancy and Reliability

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Safety Characteristics of Small Engines

The thrust levels required for SEI missions can be achieved using a few large engines or multiple small engines. It is not obvious what conclusion would be drawn from the tradeoffs for small versus large engines from an operational safety point of view. Redundancy goes up for small engines, but overall system reliability may go down.

One area where small engines have a clear safety advantage is in ground testing. Small engines will be easier and less costly to test than large engines. The test support requirements and effluent throughput will both be lower, resulting in a smaller and less costly facility. Accidents consequences will also be mitigated due to the lower fissile inventory in a small NTRE.

A detailed fault tree failure analysis will be required to determine the optimum arrangement from an overall safety point of view. This analysis should be performed as soon as possible. The tradeoffs in a safety evaluation should include certain mission parameters. For example, a lower thrust level for a longer time may result in a safety advantage. This would probably also result in a higher initial mass in low earth orbit, but this might be a worthwhile trade for increased safety.

Overall Conclusions

- The XNR2000 Uses a Demonstrated Fuel Technology Which Has Been Shown To Be Recoverable
- CERMET Fuel Has Demonstrated High Fuel Integrity and Safety Features
- At NASA SEI Conditions, Superior Fission Product Retention Expected
- There Are Ground Testing Safety Benefits To Use of Small Engines
- No Obvious Roadblocks To the Development of the XNR2000 For NASA SEI Applications Were Identified In Any of the B&W Tasks

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Overall Conclusions

B&W's overall opinion of the XNR2000 is positive. It uses a demonstrated fuel technology which has been shown to be recoverable. The CERMET fuel form has been demonstrated to have high fuel integrity and important safety features. At NASA SEI operating conditions, superior fission product retention is expected.

Ground testing considerations point to a safety advantage for small engines.

None of the B&W tasks have identified any roadblocks to the development of the XNR2000 as a viable NASA SEI NTRE.

COMPOSITE NTR ISP IS LIMITED BY BURN TIME

Thrust size	Mission burn (hr)	Design life	Composite ISP	Cermet ISP
25k	4.5	14	825	900
50k	2	8	845	900
75k	1.5	6	855	900

PRATT & WHITNEY XNR2000 CERMET NTRE

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Composite NTR Isp Is Limited By Burn Time

The performance level, specific impulse or propellant exit temperature, of the cermet based NTRE systems is not limited by burn time. However, the operating temperatures and consequently Isp, of the composite NTRE systems is limited by burn time due to the inherent chemical instability between the hot hydrogen environment and the carbon based fuel form.

ADVANTAGES OF DUAL PASS REACTOR

- **Power/Flow Matching**
- **Material Temperature Matching**
- **Flat Radial Profiles (Mixing)**
- **Isolated Hot Core**

900 seconds I_{sp}
Superior Fn/Wt

Advantages of Duel Pass Reactor

The primary attractive features provided by the dual pass reactor core are summarized. A flat radial power profile is provided by the dual-pass reactor due to the averaging of power distributions relative to two distinct regions. Positive flow/power matching is achievable because of the separation of the inner and outer cores. The maximum fuel element power shape factors appear in the outer core region because of the proximity of the radial reflector. However, because the outer core serves as the first pass, the coolest hydrogen propellant passes through the outer core and eliminates fuel temperature concern. Additionally, upper plenum mixing of the hydrogen serves to eliminate the outer core power peaks from the inner core fuel elements. The dual pass configuration isolates the hot inner core fuel elements from the rest of the engine system. This isolation provides material flexibility allowing the use of lighter weight Moly based fuel elements in the outer core and a Be radial reflector which provides the most reactivity worth for the weight. The most obvious benefit of the dual pass core is the reduced axial thermal gradients and consequently thermal stress loads placed on the fuel elements.

ADVANTAGES OF FAST SPECTRUM CERMET REACTOR

- **Safety**
  - **Positive Fuel and Fission Product Retention**
  - **Compaction/Immersion**
  - **Long Life**

- **Simple Design**
  - **No H_2 Reactivity Feedback**
  - **Simple Support (No Tie Tube Complexity)**
  - **Control Flexibility**

- **Strong High Conductivity Fuel**
  - **Self Supporting**
  - **Dimensional Stability**
  - **Resistance to Thermal/Physical Shock**

<u>Advantages of Fast Spectrum Cermet Reactor</u>

The XNR2000 builds upon the experience and database of Cermet fuels obtained in the GE710 and ANL programs. The fast spectrum Cermet fuel form was selected to meet the engine requirements of ALARA fuel and fission product release, multiple restart capability and subcriticality under credible accident scenarios. During the GE710/ANL programs the Cermet fuel form displayed tolerance to excessive temperature/power ramps due to the high strength and conductivity of the refractory metal matrix. Additionally Cermet fuel display complete compatibility in the expected hot H2 operating environment as well as cladding and fuel matrix CTE compatibility. Finally the XNR2000 is based upon a fuel form that was successfully fabricated and tested.

The selected Cermet fuel form provides a more robust, simple system design because of the elimination reactivity feedback from hydrogen moderation. A simplified support structure is possible due to the high strength of the refractory metal based fuel form.

XNR2000 CERMET NTRE NEAR TERM RECOMMENDATIONS

- 37 hole fuel element fabrication trial
  - Near net shape
  - Wavy plate
- Refine XNR2000 baseline design
  - Mechanical design
  - Transient, off nominal
  - Reliability analysis
  - Manufacturing study
  - Health monitor/control definition
- Ensure "fast spectrum" testability in PIPET

PRATT & WHITNEY XNR2000 CERMET NTRE 25824

XNR2000 Cermet NTRE Near Term Recommendations

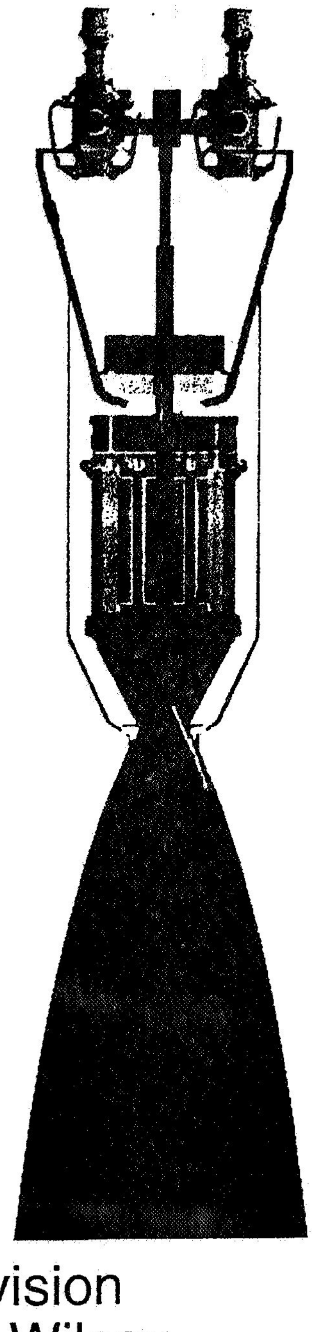
The near term development priorities of the XNR2000 Cermet based NTRE are listed. A 37 hole baseline fuel element should be fabricated using the near net shape fabrication technique with the many plate technique used as a backup. The preferred fabrication technique should be selected and refined to incorporate current powder metallurgy technology.

The baseline design effort should be continued and refined in the areas of mechanical design, XNR2000 manufacturing, and health monitor/control definition. Transient and off nominal studies of the XNR2000 are required as well as a reliability analysis.

The testability of the XNR2000 fast spectrum fuel form in PIPET should be established.

N93-26921

NTRE Extended Life Feasibility Assessment



Final Report

23 OCT 92

Presented to

NASA LeRC

By

Aerojet Propulsion Division
Energopool, Babcock & Wilcox

We Have an Effective NTRE Team

Aerojet Propulsion Division
Sacramento, California

Babcock & Wilcox Advanced Systems
Engineering
Lynchburg, Virginia

Energopool – Moscow, Russia

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NASA LeRC TOC Program Objectives

- Assess Feasibility of a Long Life, Reusable Nuclear Thermal Rocket
- Two Reactor Concepts
  - Particle Bed Reactor (PBR)
  - Commonwealth of Independent States (CIS)
- Tasks
  - Conceptual Layouts (75K lbf)
  - Thermodynamic Cycle Balance
  - Preliminary Neutronic and Thermal – Hydraulic Analysis
  - System Mass Estimates
  - Preliminary Life and Reliability Assessment
  - Safety Assessment
  - Scaling to 25 and 40K lbf (PBR Only)

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The NASA LeRC TOC Addresses the Emerging NTRE Requirements

Thrust	25K, 40K, 75K
Thrust/Wt (With Internal Shield)	> 4
Isp	> 850 sec
Length	30 Meters
Diameter	10 Meters
Throttling	25%
Restarts	> 10
Single Burn Duration	60 Min (Max)
Life	> 270 Min at Rated Thrust
Reliability	Manned

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NASA LeRC TOC Final Report Agenda

• Introduction	Wayne Dahl
• Technical Overview	Mel Bulman
• Concept Definition	Don Culver
• Engine Design	Roy Squires
• Integrated Engine	Mel Bulman
• Engine Reliability and Safety	Mel Bulman
• PBR Engine Sensitivity Study	Mel Bulman
• PBR Reactor System	Richard Rochow
• CIS Engine	Don Culver
• CIS Reactor System	Richard Rochow
• Technology Road Maps	Mel Bulman
• Summary	Mel Bulman

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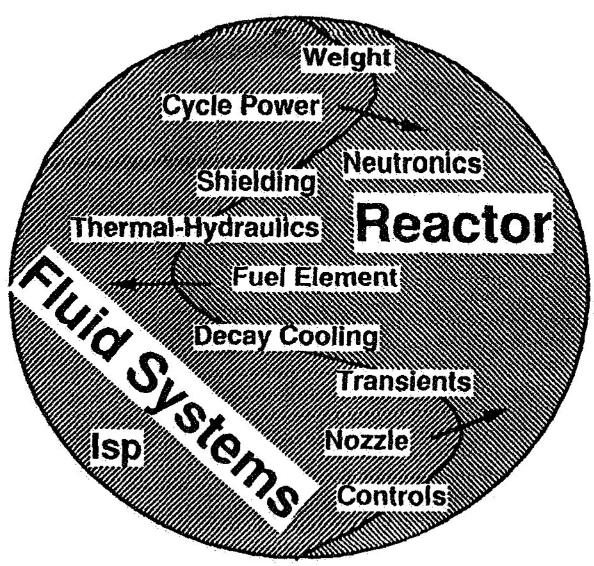
The NTRE is a highly integrated machine. As we will show, interactions between reactor and engine level operations are significant. Our systems approach to NTRE design reveals exciting new possibilities for improving the reliability and performance of spacecraft.

The NTR Engine Is a Highly Integrated Machine (Not Just a Reactor Between a Pump and Nozzle)



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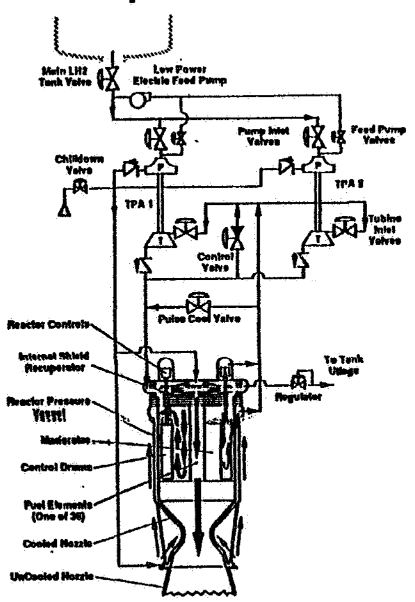
Our basic NTREs meet all current NASA requirements.

The thrust to weight ratio of the PBR engine is 6.3. The CIS engine is somewhat heavier with a F/W of 4.7. The PBR Isp is 65 seconds higher than the requirement at 915 sec. The higher temperature of the CIS engine produces an Isp of 959. Both engines fit well within the space allowed for in the SOW.

Our advanced pump design and engine management system permits throttling 20:1 compared to the requirement of 4:1.

Our preliminary life evaluation indicates the engines will be able to operate longer than currently required. Our preliminary reliability and hazards analysis indicate man rating of these engines is achievable within the scope of the engine development.

Our Basic Engine Meets All Current NASA Requirements With a Recuperated Topping Cycle



	Requirement	PBR Value	CIS Value
Thrust	75 Klbf	75 Klbf	75 Klbf
Thrust/Weight With Shield	> 4	6.3	4.6
Isp	≥ 850	915	959
Length	≤ 30M	7.7M	8.1M
Diameter	≤ 10M	2.2M	2.2M
Throttling	≥ 4:1	20:1	20:1
Reuse	≥ 10 Restarts	> 10	> 10
Single Burn	60 min	> 60 min	> 60 min
Engine Life	≥ 270 min	≥ 270 min	≥ 270 min
Reliability	Manned Stage →		

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With our recuperated cycle, we avoid complex core designs that produce heat to drive the turbopumps, yet we have increased the engine operating pressure to reduce its size and weight and to increase its performance.

We have studied two NTREs with heterogeneous reactors. One employs the particle bed reactor concept developed in the U.S. The other is based on 20+ years of development in the CIS. The CIS reactor utilizes a twisted ribbon type fuel and has been tested at over 3000K for over 1 hour.

- PBR

  In order to meet the NASA life requirement we have changed the fuel stoichiometry and lowered the operating temperature. We have arranged for deep throttling and closed loop decay heat removal.

- CIS

  We have modified our engine drive cycle and structure slightly to best make use of CIS fuel assembly technologies.

Technical Approach: Apply Our Recommended Engine Cycles to Two Heterogeneous Reactor Types

- Engine Cycle
  - Delete Gas Heater Fuel Assemblies
  - Raise Operating Pressure
  - Integrated Engine Option
- Particle Bed Reactor
  - Increase Design Life
  - Provide Deep Throttling/Decay Heat Removal
  - Integrated Engine Option
- CIS Reactor
  - Fuel Developed
  - High Operating Temperature
  - Integrated Engine Option

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There are several ways in which heterogeneous fission reactors are superior to homogeneous ones, and all result from physically separating fuel and moderator, the characteristic of the heterogeneous concept. Moderator and fuel have different requirements, and separating them allows selection of optimum solid materials for each function.

High temperature carbides are suitable for fuel, because, when used correctly, they can deliver high reactor gas outlet temperatures, which enables high engine specific impulse. High gas temperatures are available, because carbide fuel can operate at high temperatures and because if formed into thin elements, internally generated heat need pass only a small distance to the coolant. Thus, it need not pass through moderator material to reach a cooled surface, as in the homogeneous reactor concept. Propellant gas can attain a temperature very close to the fuel's maximum internal temperature. The PBR attains this advantage by using small diameter spheres in fuel particle beds, while the CIS reactor uses bundles of thin, twisted ribbons of fuel.

Efficient neutron moderators are hydrogenous, and no solid materials of this type can withstand temperatures in the range of fuel or desired outlet gas temperatures. Efficient neutron moderators are important, because uranium fission cross-sections are very low at fission neutron energy levels, and without good moderator material a larger amount of fissionable material is needed in the reactor. Several negative features occur simultaneously when large amounts of highly enriched U235 are used in a reactor. Primarily, the safeguards problem is worsened. Secondly, launch safety is inherently less. Third, fast reactors need a more rapid control system, which exacerbates development and safety risks, and fourth, fuel cost is much greater than that of the moderator which may replace it in a heterogenseous reactor. The PBR moderator is hexagonal blocks of beryllium containing cavities filled with LiH that surround each fuel bed, while the CIS moderator is ZrH2 rods close-packed between the fuel assemblies.

For Mars mission NTRE we need a specific-impulse-loss-free turbopump power cycle to minimize total mission costs, including Earth-to-orbit launch. Thus, topping cycles are used, which have turbine life advantages over bleed cycles. In a heterogeneous engine the lower temperature moderator and reflector materials are cooled with a separate hydrogen loop prior to final heating by the fuel elements. This moderator and reflector heat is automatically the major portion of the topping heat needed for turbine inlet gas heating – for turbine drive power. In a homogeneous engine, at least the moderator heat is lost for turbine drive use. Lower engine operating pressure results, all other things being equal, and this leads to large, heavy engines with inferior Mars mission performance. Further, the moderator cooling loop also enables integration of a closed engine cooling and electric power generating system that can reduce Mars mission IMLEO by about 100 tons.

Heterogeneous Reactors Superior to Homogeneous Types (NERVA)

Features	PBR	CIS	Benefits
• Fuel Separated from Moderator			• Moderator Cooling Powers Turbine and Enables Closed Loop Cooling (Reliability and Weight)
• Fuel More Efficient	Spheres	Twisted Ribbon	• Higher Gas Outlet Temperature (High Isp)
• Moderator More Efficient	Be Hex with ZrH_2 Cavities	ZrH_2 Rods	• Lower Fissile Inventory (Safety and Weight)

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17 69 5cAv8

Our basic engine meets or exceeds all NASA requirements. It provides for robust operation and takes up little room in the launch vehicle.

As we studied these engines we recognized some significant and beneficial differences between these heterogenously moderated reactors and the homogeneously moderated NERVA type reactors. The separate moderator allows us to extract significant heat from the core without the need to flow hydrogen through the fuel elements. The full utilization of this in our integrated engine provides many benefits including: (1) reliable, efficient NTRE start up, (2) reduced decay cooling losses; (3) RCS and OMS at high Isp, (4) electrical power up to 100 kW (E) per engine.

Technical Approach: Two Engine Options Are Presented

- Basic Engine
  - Meets or Exceeds All Current NASA Specs
  - Robust Operation
  - Reliable, Efficient Engine Starting
  - Small Size
- Integrated Engine
  - Builds on Basic Engine
  - Reduces Decay Cooling Losses
  - Improves Mission Reliability and Performance by:

    Integrating Stage and Engine Subsystem
    Main Propulsion
    RCS
    OMS
    Option for Electric Power (~ 100 kWe)

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1789 Sc/1/9

Our platelet technology enables us to turn the requirement to cool the internal gamma shield into a cycle-enhancing recuperator without mass penalty. This allows us to operate the engine at higher chamber pressures than otherwise possible, resulting in a smaller and lighter weight engine. In addition, the recuperator provides the bulk of the energy for the engine start. Sufficient energy is stored in the recuperator to accelerate the turbopumps to full power without additional heat. With this magnitude of stored energy, it would take over 10 aborted starts to significantly reduce the starting power of our cycle.

In addition to providing power, the recuperator provides thermal and hydraulic stability during all modes of engine operation. The reactor and feed system are effectively decoupled during high reactor transients.

Recuperated Cycle Provides Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
  - Energy Available for Many Starts
- Provides Safe, Controllable Reactor Start
  - Prevents Liquid Hydrogen Entry Into the Core
  - Decouples and Damps System Oscillations

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Engine Concept Definition

Don Culver

Our engine and major component design concepts are selected to meet all current NASA requirements. Any concepts that cannot meet these safety and reliability, performance, and operational requirements in the near term were discarded. In addition, many NASA goals that impact safety and reliability, mission benefits, development cost and technical risk were used to guide system configuration selections and design and operating parameter optimization studies.

NASA Goals and Requirements Impact APD NTRE Selection

Requirements

- Safety
  - Radiation Protection
  - Manrate, Verify, Automate
- Performance
  - 850 sec Isp
  - 4:1 Thrust Weight
  - Throttling @ Tmax
  - 15-75K lbf Thrust
- Operation
  - Reusable, Long Life
  - Bootstrap Start w/o Power
  - Degraded/Failed Tolerance

Goals

- Safety
  - Minimize Radioactive Materials
  Hazard Mitigation and Reliability
- Mission Benefit
  - IMLEO/Trip Time (Isp and F/W)
    Mission Commonality
  - 2006 Availability
  - Simplicity (Inherent Reliability)
- Technical Risk and Development Cost
  - Technology Readiness and Tests Needed
  - Propulsion System Integration
  - Facility Requirements

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17.89 Sch/172

We begin our concept definition studies with trade studies. Of course, we perform trade studies in each important area of requirements and goals.

When trade studies are completed, optimum design parameters are known, and engine layout and component design studies can be finalized. When the point design is known, sensitivity studies are made to check the impact of important design and operating parameters on engine characteristics.

Trade Studies Define Engine Concept and Design Point

• Safety and Reliability	
– Nuclear	Criticality Trades (B&W)
– Non Nuclear	Feed System Reliability
• Performance and Mission Benefit	
– Mission Payload	Versus Cycle Type, Pc, Nozzle Design
– Power Cycle	Definition (Shield Integration)
– Control System	Architecture Study
• Operation and Technical Risk	
– System Operation	Modes and Procedures Identified
– Propulsion System Integration	Shield, Decay Heat, Deep Throttling
– Technology Readiness	Major Component Status

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Our Reliability Plan Is Tailored to Project Phase

- Concept Phase (TOC)
  - Reliability Block Diagrams With Typical Component Failure Rates
  - Preliminary FMEA to Component Level
  - Hazards Analysis (Crew, Ground Support and Populace)
- Design Phase
  - FMEA
  - Fault Tree Analysis
  - Safety Studies

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17.69-5cm/14

A result of our feed system reliability block diagram study is that use of twin turbopumps on NTRE should improve mission reliability by reducing the probability of total failure and engine loss to about 1/4 of that of single turbopump fed engine. However, the twin turbopump engine has nearly twice the probability of failing to a degraded mode of performance. This usually means that one turbopump fails and the other continues to operate the engine at nearly 3/4 thrust. This is of little consequence at any time except a TMI (or TLI) burn.

Twin Turbopumps Improve Mission Reliability*

- Single TPA System Has ~ 4 Times the Probability of Total Failure vs 2 TPAs
- Twin TPA System Has ~ 1.7 Times the Probability of Failure to Degraded Mode (~ 70% Thrust) vs 1 TPA

* Industry Standard Component Failure Rates Applied to Feed Systems

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Mission performance depends on rocket engine thrust/weight and mission average specific impulse (Isp). Engine thrust/weight depends largely on reactor type and power density, engine configuration, and operating conditions. Mission average Isp depends mainly on engine Isp and on operational Isp losses, and they depend on mission type, engine design details, and operating conditions. We will discuss our trade study results for each of these factors in the following charts and in the reactor design sections.

Mission Isp Depends on Engine Isp and Operational H_2 Losses

- Engine Isp = f $(Tout)^{1/2}$
  - Theoretical Isp (Tout and ε)
  - Tout max – Tout Mixed Mean
  - Nozzle Losses (Cooling, Divergence)
  - Power Cycle Bleed Losses
- Operational H_2 Losses
  - Open Loop Cooldown @ T < Tmax
  - Boiloff and Leaks
  - Start-up Bleed

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We have studied three fundamental engine configurations:

(1) DeLaval nozzle behind thermal reactor

(2) Forward flow thermal reactor within an expansion-deflection (E-D) nozzle

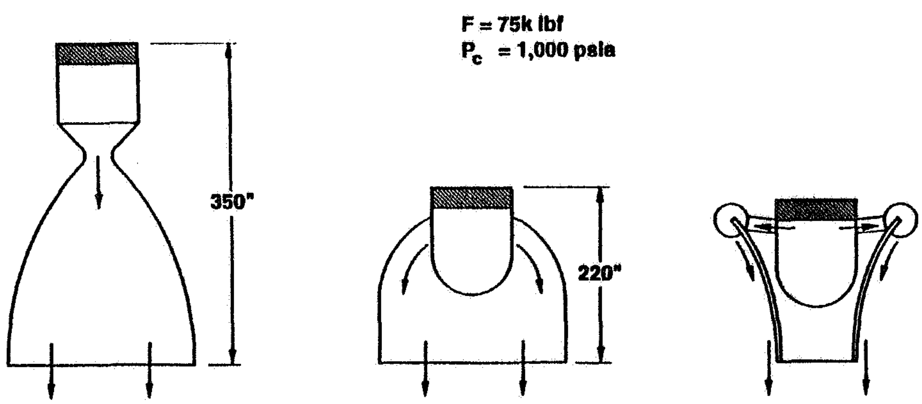
(3) Forward flow thermal reactor within a plug nozzle

The E-D nozzled engine appears to have the best mission performance potential, but it needs further study, and at this time it is recommended for a second-generation engine. However, we recommend this concept be studied in more detail soon, because it is rapidly developing into a more practical concept than was believed possible earlier.

The plug nozzled engine does not seem competitive, because of its large nozzle surface area in the high heat flux region of the throat and its consequent low Isp and high weight potentials.

The DeLaval nozzled design is, thereby, recommended for a near-term engine.

DeLaval Nozzle Is Attractive for Near Term NTRE



DeLaval	E/D	Plug
• Low Losses	• Good Integration	• Highest Nozzle Heat Flux and Losses
• Long and Heavy	• Short and Lightweight	• Short

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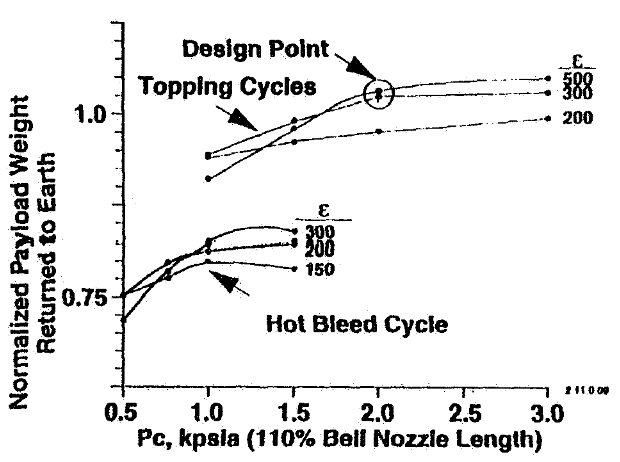
A trade study evaluated both engine weight and specific impulse by estimating the Mars mission payload delivery capability of identical vehicles powered by similar engines of conventional geometry having different power cycles, operating pressures, and nozzle area ratios. Both hot bleed cycle engines and topping cycle engines were evaluated over reactor outlet pressures (Pc) from 1000 to 3000 psia, with nozzle area ratios from 150 to 500, and with nozzle lengths from 80 to 120 percent bell. In each case a cooled, copper and steel nozzle was used with a carbon-carbon nozzle extension from area ratio 10 to the exit.

Results showed that bleed cycle engines are not competitive, based on their lower delivered specific impulse. Their payload carrying capabilities were consistently low by about 20 percent. Nozzle contours of 110 percent bell length were found to be best for nearly all engine variants. Engines with high nozzle area ratios benefited most from high engine pressure, because their nozzles are smaller and lighter in weight, better offsetting the increased turbopump weight required of high pressure feed systems. Conversely, engines with low nozzle area ratios are relatively insensitive to engine design pressure. (Both reactor design teams agreed that reactor, vessel, and shield weight totals are not greatly affected by design pressure in the range of our study.)

The design point selected was area ratio 300 with pressure of 2000 psia, because it appeared to be the lowest pressure – lowest area ratio combination to attain high mission performance. At 200 nozzle area ratio about five percent payload is lost, regardless of engine pressure selection.

High Pressure Topping Provides Maximum Mission* Performance

Study Results



Engine Selection

- Topping Cycle
- Pc = 2,000 psia
- ε = 300 (De = 92 in.)
- L_{noz} = 110% Bell (210)
- Tc = 2,700K

* 4 Burn, All-Up, Manned Mars Mission With C_3 = 16 km²/s² and IMLEO = 775 Tonnes

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We examined all reasonable turbopump drive power cycles, based on examination of heat sources for turbine inlet gas and destinations for turbine exhaust gases. Those cycles which bleed turbine exhaust gas overboard rather than through the throat of the engine's nozzle lose specific impulse, because they cannot expand low pressure and temperature turbine exhaust gas to high velocities. We found that these engines are not in contention for Mars missions on a performance basis.

Three topping cycles were analyzed. The simple expander cycle cannot provide enough heat to power the engine reliably to pressures of 1000 psia or above. We have seen the Mars mission performance decrement of low pressure engines. Therefore, extra heat must be added to the topping heat that can be recovered indirectly from the reactor core by cooling engine components, such as moderator, reflector, pressure vessel, shields, etc. One way to do this is to devote a portion of the fuel assemblies to turbine drive heat. This requires additional manifolding in the reactor core, an unwanted complexity which may reduce neutronic efficiency and cost engine size and weight. Another way is to use a high heat rate heat exchanger to transfer turbine exhaust heat to the pump discharge to augment the topping cycle heat. This is the scheme we selected, in spite of the fact that engine designers usually feel that highly effective recuperators are large and heavy.

Recuperated Expander Cycle Selected

Turbine Exhaust Destination	Heat Sources for Turbine		
	Reactor Core	System (Expander)	
		Recuperated	Not Recuperated
Overboard = Bleed Cycles	Hot Bleed	Recup. Bleed	Cold Bleed
• Isp Loss • Partial Admission • Large Turbine	• Hot Turbine • Nozzle Port • Mixer Fatigue • ≈20% P/L Loss	Similar to Cold Bleed	• Larger Toploss or • Hot Turbine • ≥14% P/L Loss
Reactor = Topping Cycles	Augmented	Recuperated	Expander
• More Valves	• Core Complexity • Reactor Size and Weight	• Recuperators Typically Large and Heavy • Max. P/L	• Limited Power • Restart Heat? • 7% P/L Loss

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The impacts of the recuperator on our engines' size and weight is nil, because:

(1) We have demonstrated our ability to fabricate large heat rate heat exchangers of very compact dimensions with our platelet technology, for example in the SSME heat exchanger program.

(2) The steel recuperator can function as the gamma shield for the NTRE and the forward closure of the reactor vessel. We have shown that the sum of these two weights in a conventionally designed engine are greater than the required recuperator weight. Thus, we incur no weight penalty for the heat exchanger itself.

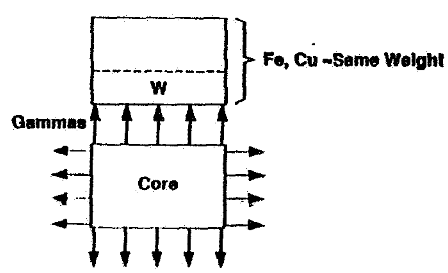
(3) Low density material, such as steel may be used efficiently for a gamma shadow shield, because it is located close to the large diameter reactor, and the radiation tends to be planar to all surfaces, because of the self shielding provided at all other angles.

Recuperator Weight Impact Is Nil

Problem – Large Recuperator Size and Weight

Solution – Compact Stainless Steel Platelet HEX Doubles as Cooled Internal Gamma Shield and Forward Pressure Vessel Head

Distributed Source Shield Weight Is Not Dependent on Material Density



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Recuperated Cycle Provides Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
  - Energy Available for Many Starts
- Provides Safe, Controllable Reactor Start
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  - Decouples and Damps System Oscillations

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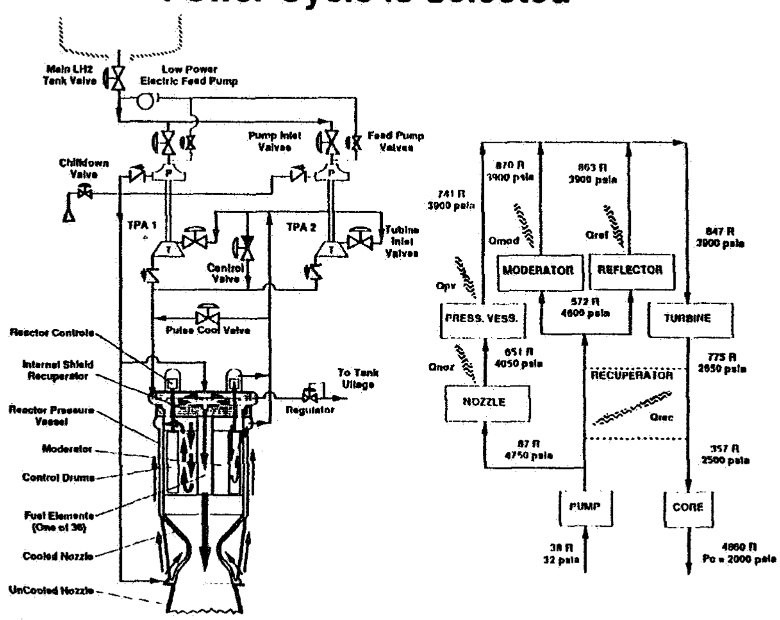
The power cycle we have selected for both engines uses a recuperator to transfer heat from turbine exhaust to pump discharge, and each cool the nozzle with a small side stream of liquid hydrogen from the pump. The PBR cycle variation is shown here; it requires a pump discharge pressure of 4750 psia to deliver an engine Pc of 2,000 psia. It does this with a low turbine inlet temperature of 847 R (470 K) and a low turbine pressure ratio of less than 1.5 to 1. Pump stage pressure ratios are low, too, because four stages of pumping are used. However, four stage rotating assemblies are not needed with our concept, because our turbopumps emulate a quad-redundant valve set. We use two turbopumps in parallel to provide the total engine flow, and each turbopump consists of two identical rotating assemblies operating in series. Each rotating assembly is the simplest configuration possible, two pumps and a turbine on the shaft with two bearings between the three rotors. Reliability, performance, risk, and cost benefits result from this subcritical speed design.

The recuperator heat rate is about 125,000 Btu/sec, which is larger than the sum of the topping heat. If more power is needed this cycle has two main design variables, turbine pressure ratio and recuperator heat rate. The latter controls the turbine inlet temperature. The power balance shown has ten percent excess turbine power for turbine bypass control authority.

The flow scheme through the engine is as follows: through the pumps in parallel, with a 5-1 flow split after their flows join; the small flow cools the nozzle and pressure vessel; the large flow gets heated in the high pressure side of the radial outflow recuperator, where it enters the moderator and reflector cooling flow at the front of the core; the full flow passes through the turbines in parallel to rejoin and cool in the low pressure, radial inflow passages of the recuperator; cooled flow is manifolded to the inlets of each fuel assembly for heating to full outlet temperature and passage through the rocket nozzle.

Flow control elements include a low power electric feed pump, pump and turbine inlet and outlet valves, a turbine bypass control valve, and a pulse cooling valve. Reflector drive motor shafts penetrate the recuperator at its periphery, outside of the heat exchanger region, and a launch poison rod penetrates it at its center.

A High-Power, Loss Free Engine Power Cycle is Selected



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The basic engine has five operating states in addition to two additional control states, checkout and emergency. Before starting the engine, the pumps are chilled with tank head or feed pump flow, and GH_2 is vented overboard as required. However, much of the GH_2 is pumped under pressure into the engine power loop by the feed pump. The engine can be held in this stage of chilled and pressurized readiness for long periods, with occasional chill down flows until impulse is needed.

Starting the engine consists of opening the turbine inlet valves and blowing down the loop, spin starting the turbines. The large, available amount of sensible heat in the recuperator bootstraps the feed system until reactor heat is available. The recuperator prevents liquid hydrogen from ever entering the reactor.

During engine operation at high power the engine thrust is controlled with turbine bypass valve position and specific impulse with reactor control drum position. The propellant tank is pressurized by a bleed from the low pressure recuperator outlet manifold.

Following reactor shutdown with control drum rotations, the 10-1 throttling turbopumps are throttled to maintain outlet temperature by their bypass control valves. When they have reached their minimum flow, one turbopump is shut down and the other (throttled up 2-1) will follow the reactor power down to about five percent and then begin to overcool the core, reducing specific impulse at this low thrust level. The electromechanical feed pump is started, and propellant is pumped under pressure at low flow rate into the cooling loop. When the loop pressure is high and the core cool, the second turbopump is shut down and the pulse cooling valve actuated. The core heats during pump shutdown and overcools during the cooling pulse. The pulse valve shuts, the feed pump pressurizes the loop while the core heats, and the valve cycles again, holding the average core outlet temperature and Isp above what it would be without pulse cooling.

While the core power decays the duty cycle of the pulse cooling valve changes continually, and eventually it stays closed. This happens when the pressure vessel is able to radiate the residual core afterheat to space.

Operation Features Robust Start and Efficient Cooldown

- Readiness
  - Pressurize Loop With Feed Pump
  - Chill Pumps and Vent GH_2
- Start
  - Blowdown Start TPAs With Start Valves
  - Bootstrap on Recuperator and Reactor Heat
- Run
  - Control Valve Throttling
  - Bleed-Pressurize Tank Ullage
- Cooldown
  - Shutdown Reactor
  - Throttle on Decay Heat
  - Shutdown 1 TPA and Throttle to 5%
  - Overcool Fuel and Start Feed Pump
  - Shutdown 2nd TPA and Pulse Cool
- Soakout
  - Stop Pulse Cooling and Radiate

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Recuperated Cycle Provides Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
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  - Decouples and Damps System Oscillations

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Engine Design

Roy Squires

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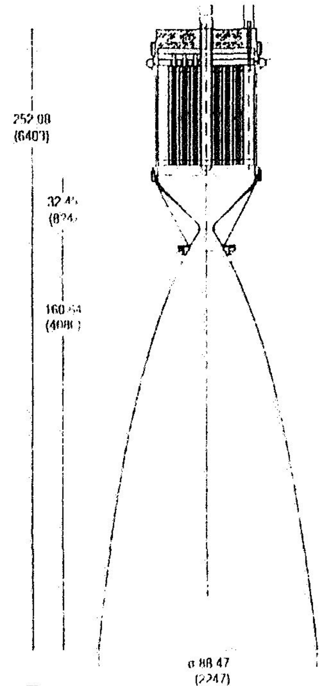
Aerojet NTRE Is Small and Lightweight



	PBR
Thrust, lbf	75000
Chamber Pressure, psia	2000
Nozzle Area Ratio, Ae/At	300
Engine Specific Impulse, sec	915
Mars Mission Specific Impulse, sec	887
Engine Total Weight, lbm	11879
Thrust/Weight	6.3
Engines per Vehicle	2
Payload Returned to Earth, lbm	44900

Engine Weight Breakdown

Component	
Uncooled Nozzle	240
Cooled Nozzle	1000
Pressure Vessel, Reactor Manifolds & CSS	1501
Reactor, Reactor I&C	3982
Turbopump Assemblies (2)	410
Recuperator / Shield	2168
Secondary Shield	521
Plumbing/Valves	1320
Controls and Shielding	737

NTRE w/ Stage Power/Heat Removal

Stage Power & Heat Removal Sys Wt, lbm	2000
Engine with Power Sys Wt, lbm	13879
Mars Mission Specific Impulse, sec	905
Payload Returned to Earth, lbm	50226

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Aerojet NTRE

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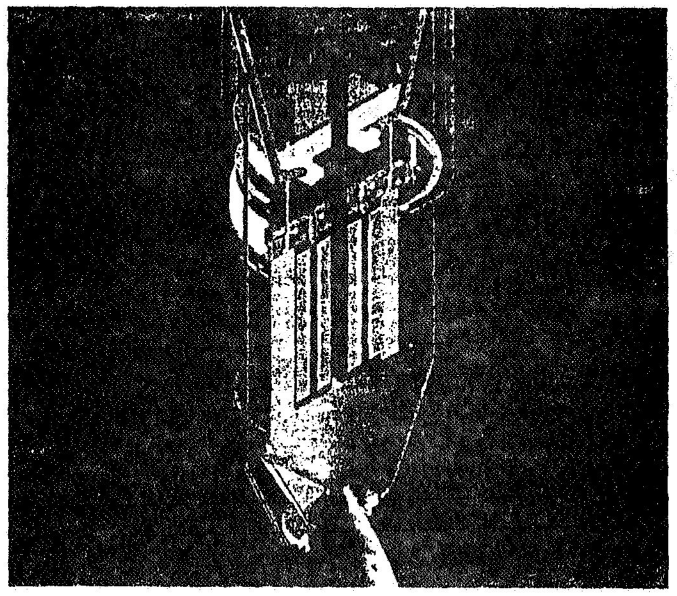
Aerojet NTRE



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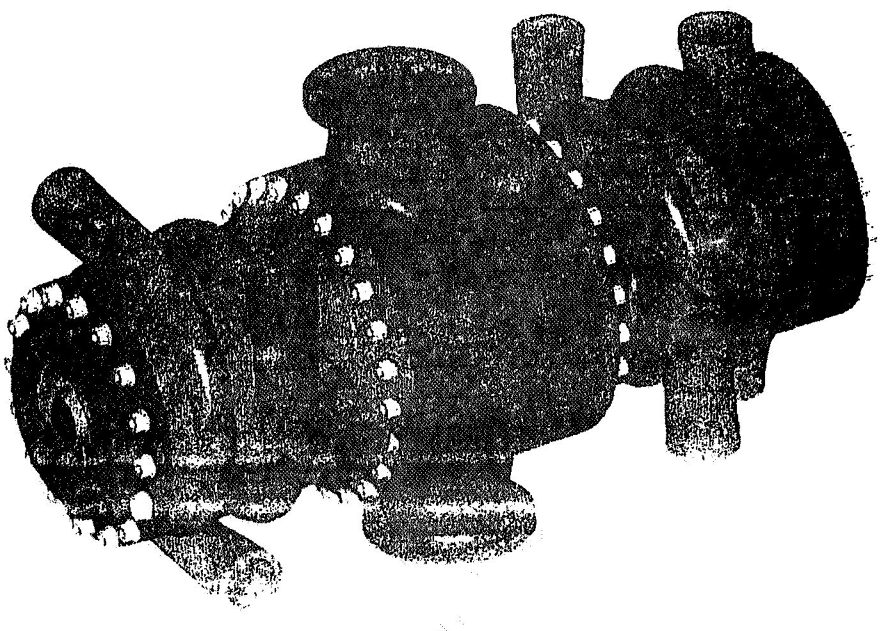
CAD TPA Dual Spool 3-D Color Exterior View



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XLR-134 LH_2 TPA

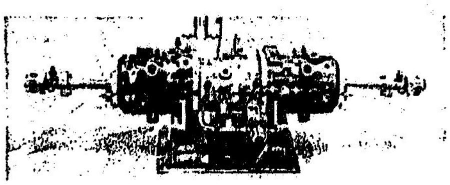
The XLR-134 program addressed the need for a relatively low thrust engine to move large fragile structures from low earth orbit into higher orbits. It was an Air Force program originating from the Phillips Lab at Edwards Air Force Base and spanned 1986 through 1990.

The program included initial studies to define the requirements and the engine size/cycle. From these requirements the engine and component designs were derived. The selected engine was a 500 lbf. LOX/LH_2 single expander cycle engine (gaseous hydrogen turbine drive). The turbopumps for both the LOX and LH_2 were designed and fabricated at Aerojet. The general arrangement consists of two shafts with 3 pump stages and one turbine stage on each mounted "end to end." In this configuration the turbines are counter-rotating. The LOX TPA is basically a two stage single spool machine of a similar design as the LH_2 TPA with appropriate material and tolerance changes.

The LH_2 TPA was tested both as a single spool (3 stage) TPA and finally as the complete dual spool TPA. No development problems were encountered, due to the robust design and subcritical shaft speed. Of significant merit during dual spool testing was the start up and steady state operation of the two pump spools. This highly successful testing demonstrated over 4200 seconds of run time in LH_2 with full speed TPA operation, speed tracking of the two spools, successful bearing performance and subcritical shaft speed.

Aerojet TPA Technology Increases Life and Reliability

Features	Benefits
• Dual Spool	• High Turbine Efficiency • Low Weight • Commonality of Parts
– Short Shafts With 3 Rotors per	• Subcritical Shaft Speed Operation for Deep Throttling
• Operate Below Design Speed	• Increased Life and Reliability
• Hydrostatic Bearings	• Increased Life

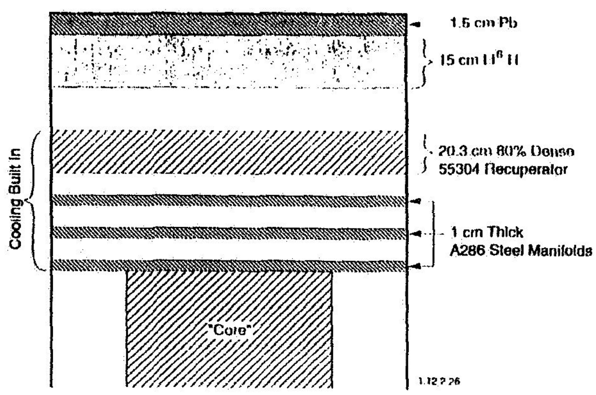


XLR-134 Fuel TPA

Radiation transmission through manifolds includes 36 2.5 cm holes for gas flow. Shield penetrations for drum control rods and the central "poison" rod were ignored.

Heating in the Li_6H/Pb dedicated shield will be of the order 40-60 kW and may require some cooling during extended operation at full power.

"Internal Shield" Concept for NTRE Provides Significant Reduction in Accountable Shielding Mass



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Source strength and shield attenuation calculated by B&W using MCNP (Monte Carlo Neutron Photon transport code).

NASA radiation specification met or exceeded at a point 1 meter above the top reflector on the core axis.

Shields for electronics and controls assumes optimum placement and 100K-rad hardened electronics.

Engine Components and Dedicated Shielding Attenuate Radiation to Meet NASA Requirements and Protect Electronics and Controls

Components	Gamma Factor	Fast Neutron Factor	Mass (Kg)	Comments
Gas Manifolds and Recuperator	101	21	1178	Dual Function: Cools and Shields
Dedicated Shield	4.4	80	236	Additional Shield Necessary to Meet NASA Spec
Distributed Electronics and Controls Shield	6.3×10^4	1.75×10^3	130	Required Beyond NASA Spec for 3.5 Hours at Full Power

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C-4

Recuperator

Function

- Internal gamma shield cooled by LH_2
- Provides thermal energy for starting and operating TPA
- Enables high chamber pressure for lightweight, compact NTRE

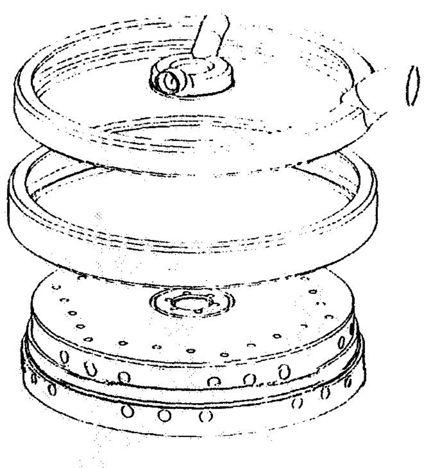
Design and Performance Parameters

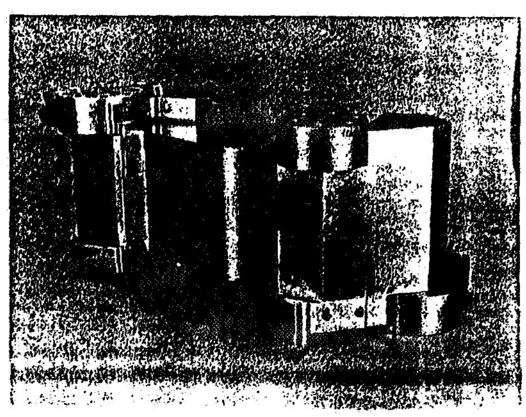
Propellant	H_2
Cold-Side Inlet Pressure	4750 psia
Cold-Side Inlet Temperature	87°R
Cold-Side Flow Rate	68 lbm/sec
Cold-Side Pressure Drop	150 psid
Hot-Side Inlet Pressure	2650 psia
Hot-Side Inlet Temperature	775°R
Hot-Side Pressure Drop	150 psid
Thermal Load	126,000 Btu/sec
Envelope	40 in. dia x 7 in. height
Weight	2500 lbm
Material	CRES SS (A-286)

Characteristics

The 300 series stainless steel, platelet design, counterflow HEX accepts 83% of the LH_2 flow from the TPAs and heats the hydrogen to 572°R gas in the high pressure circuit of the HEX. The outflow cools the reflector and moderator, ensuring that LH_2 does not enter these components. This gas is combined with the 17% flow, which bypassed the HEX to cool the nozzle and pressure vessel and was gasified in the process, to provide 100% flow at 847°R to drive the turbine. The turbine effluent then passes through the low pressure circuit of the HEX giving up much of its heat to the high pressure circuit before delivery to the reactors many fuel elements.

NTRE Recuperator Is Based on Aerojet SSME HEX Technology





SSME HEX

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Cooled Nozzle

Function

- Provides DeLaval nozzle entrance section and exit to area ratio 10:1

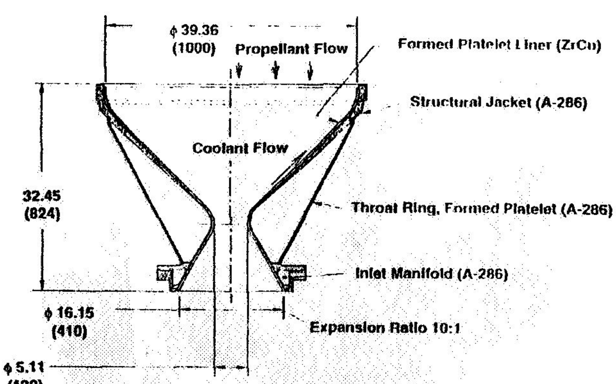
Design and Performance Parameters

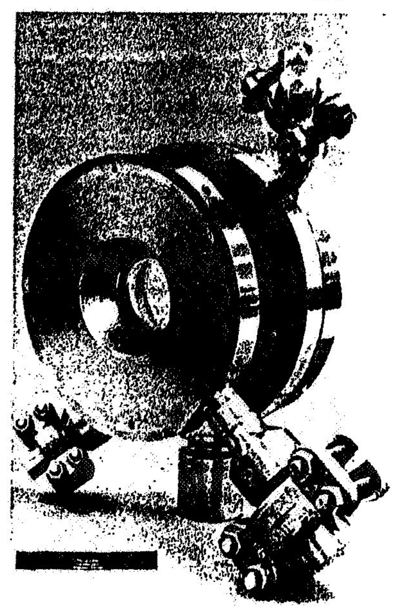
Propellant	H_2
Coolant Inlet Temperature	87°R
Coolant Inlet Pressure	4750 psia
Coolant Flow Rate	14 lbm/sec
Coolant Pressure Drop	700 psia
Throat Diameter	5.121 in.
Exit Area Ratio	10:1
Chamber Pressure	2000 psia
Gas Temperature	4860°R
Flowrate	82 lbm/sec
Material	ZrCu Liner/A286 SS Structure
Envelope	40 in. dia x 32 in. long
Weight	1000 lbm
Cylindrical Length	5.00 in.
Wall Temperature	600 F

Characteristics

The cooled nozzle uses a zirconium/copper, formed platelet liner to maintain wall temperature below the life limit. The liner will consist of 8 to 10 panels and include an approximate total of 400 coolant channels. It is bonded to a two-piece, A-286 jacket by a hot isostatic press (HIP) process. A two-piece, formed platelet A-286 throat stiffening shell provides structural support against bending moments. Its construction and cooling approach is similar to that of the pressure vessel shell. Coolant enters a manifold at the 10:1 area ratio and flows forward through the liner and shell wall as shown. It exits into the aft closure ring manifold of the pressure vessel.

Cooled Nozzle Concept Is Based on Current Technology





Formed Platelet Liner 40Klbf Chamber

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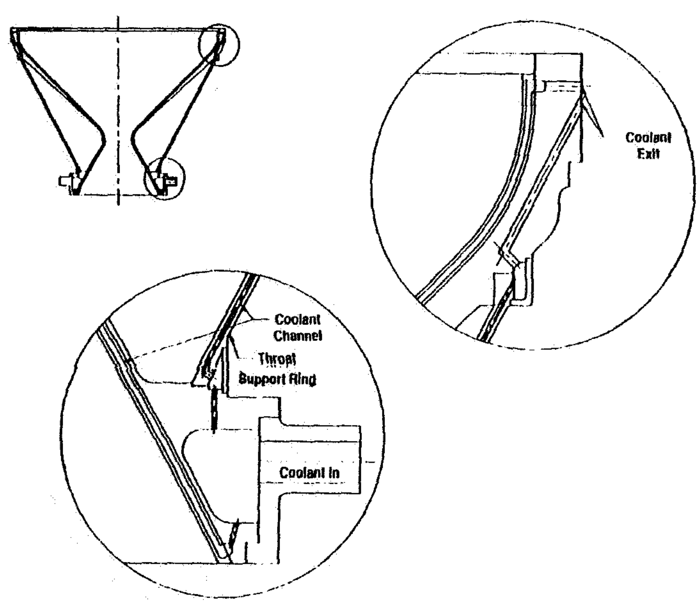
Cooled Nozzle Concept

- Studies of the SSME main combustion chamber show wall temperature reductions of up to 200 F using platelet liner technology.
- Cooled hot gas wall
  - Formed platelet liner
  - ZrCu platelets
  - ~400 channels
  - 8-10 panels
- A-286 structure
- Cooled throat support ring
  - Platelet A-286 structure with internal coolant channels formed into conical shape

Common Manifolding Provides Coolant for Nozzle and Throat Support Ring



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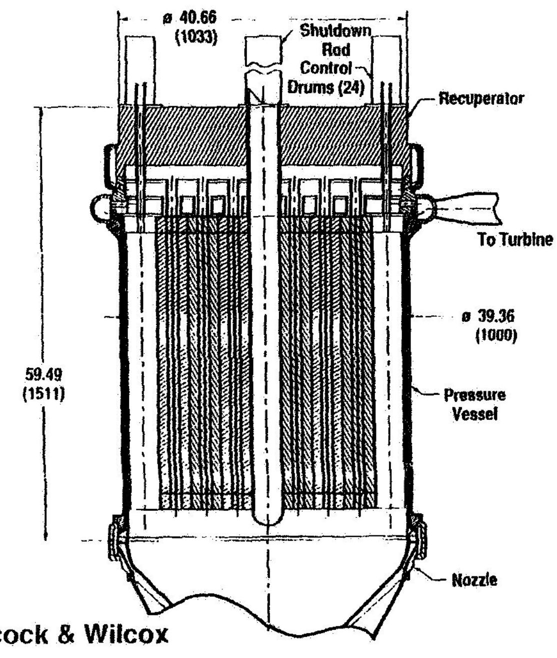
Pressure Vessel Function / Concept

- Pressure Containment
- Plenums / Manifolding for:
  - Moderator coolant
  - Control drum coolant
  - Core flow
  - Pressure vessel wall
- Interfaces
  - Recuperator
  - Cooled nozzle
  - Reactor
  - Core support

Pressure Vessel Provides Pressure Containment, Core Support and Manifolding

- Cooled Hot Gas Wall
  - Formed Platelet Design
  - A-286 Stainless
  - 3-4 Sections



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Pressure Vessel (PV)

Function

- **Contains pressure and supports reactor**
- **Provides manifolding for GH_2**
  - **Directs recuperator cold flow to control drum and moderator/reflector outflow**
  - **Combines nozzle/PV coolant outflow with moderator/reflector outflow for delivery to turbine**
  - **Delivers recuperator warm flow to reactor heating elements**

Design & Performance Parameters

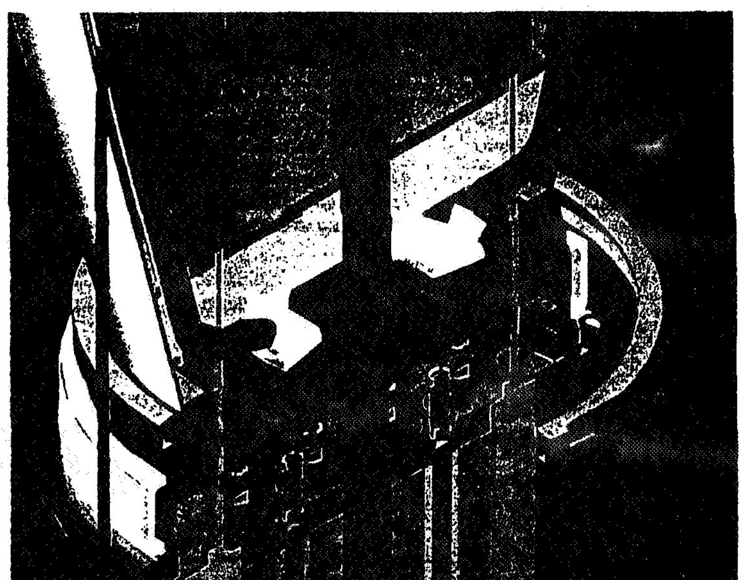
Propellant	H_2	Moderator Coolant Temperature	572°R
Coolant Inlet Temperature	651°R	Moderator Coolant Pressure	4600 psia
Coolant Inlet Pressure	4050 psia	Moderator Coolant Flowrate	53.5 lbm/sec
Coolant Outlet Temperature	847°R	Reflector Coolant Temperature	572°R
Coolant Flowrate	14 lbm/sec	Reflector Coolant Pressure	4600 psia
Coolant Pressure Drop	150 psid	Reflector Coolant Flowrate	14.5 lbm/sec
Core Propellant Temperature	357°R	Envelope	40 in. dia x 53 in. long
Core Propellant Pressure	2500 psia	Weight	2610 lbm
Core Propellant Flowrate	82 lbm/sec	Material	A286 SS

Characteristics

Formed, A-286, diffusion bonded platelet wall sections are welded together to make the right circular shell of the pressure vessel. The forward end of the shell is welded to a manifold assembly, which is welded to the recuperator. A coolant ring manifold is welded to the aft end of the shell. Annular closure rings are welded fore and aft as shown to combine flows per the engine system schematic.

Hydrogen gas enters the aft end manifold from the cooled nozzle. It flows through the shell coolant passages, etched into the wall platelets, and into a forward closure ring, where it mixes with the moderator/reflector coolant outflow for delivery to the TPA turbines. The foremost closure ring delivers recuperator flow to the moderator/reflector coolant passages.

Core Support Structure Provides Reactor Manifolding



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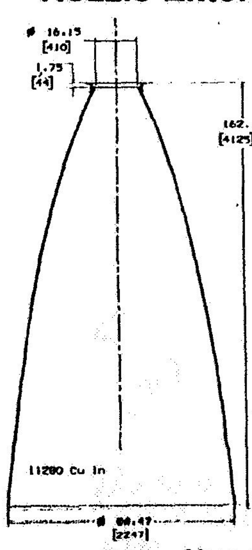
A carbon-carbon nozzle extension for NTRE is a cost effective, low weight component. Carbon-carbon has good mechanical properties for temperatures in excess of 5000°F and will only suffer a total recession of less than 0.005 inch due to hydrogen chemical attack (assuming a temperature of 2500°F, pressure of 20 psia, and total duration of 4.5 hrs). Carbon-carbon is noted for radiation resistance and was baselined as the nozzle extension for the NERVA rocket engine at Aerojet.

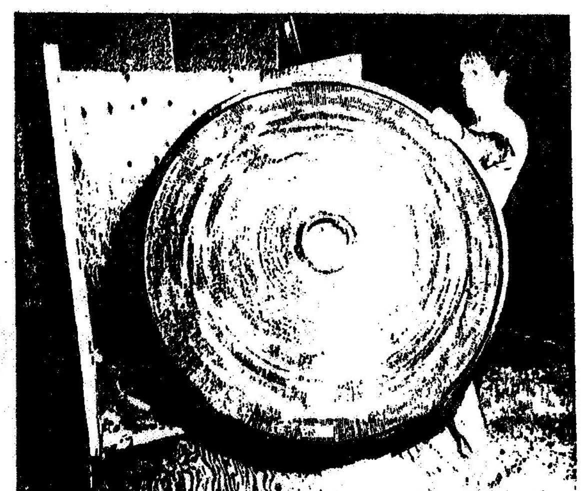
Carbon-carbon structures can be fabricated in many different ways, but only several are appropriate for thin wall nozzle extensions. Involute, 3-D cylindrical, braided, and Novoltex™ preforming are the four most realistic techniques to provide carbon-carbon nozzle extensions. None of these techniques can provide a single piece nozzle the size required without facility capitalization and development.

A one-piece carbon-carbon nozzle extension is estimated to weigh about 170 lbs and 240 lbs for area ratios nozzles of 200:1 and 300:1, respectively. The thicknesses reflect minimum wall thicknesses of approximately 0.5 in. and 0.2 inch for the entry and exit regions.

Propellant	H2
Temperature	4860°R
Flowrate	82 lbm/sec
Attach Area Ratio	10:1
Exit Area Ratio	200:1
Nozzle Shape	110% Bell
Material	Carbon/Carbon
Envelope	88.7 in. dia x 160 in. long
Weight	450 lbm

A Full Size One Piece Carbon-Carbon Nozzle Extension Will Weigh Less Than 450 lbs





However, Facilities Must Be Upgraded for Size and Nozzle Fabrication Must be Validated

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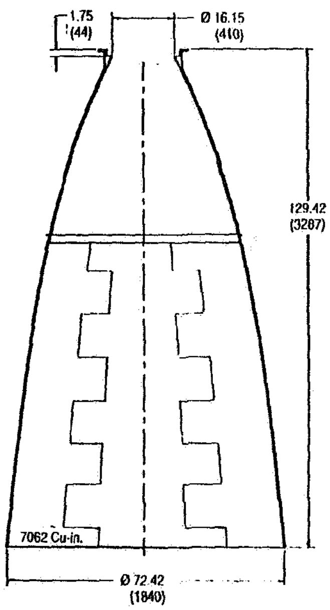
Instead of fabricating a one-piece carbon-carbon nozzle extension, fabricating carbon-carbon segments is an option which will not require facilitization nor extensive validation. A defect in a large one-piece carbon-carbon nozzle would cause the rejection of the whole nozzle or acceptance of materials of lower mechanical properties, while an unacceptable nozzle segment will only require the rejection of that one segment. The segmented carbon-carbon nozzle extension concept shortens the design and fabrication cycle by at least one year.

Aerojet has pursued the segmented nozzle approach under IRAD and has validated the mechanical approach via demonstration aluminum and fiberglass epoxy segments. The main drawback corresponds to the thickened sections in the nozzle to effect the segments attachment. The segmentation approach is estimated to increase the weight 30%.

A Segmented Carbon-Carbon Nozzle, Though Heavier, Is Robust and Cost Effective



- Fabricability
  - Present Facilities Are Large Enough to Produce Required Segmented Pieces
  - Lower Rejectable (Only Bad Segments Need to Be Replaced, Not Whole Nozzle)
- Compactness
  - Disassembled Pieces Are Easy to Store, Ship, and Reassemble
- Robustness
  - Smaller Pieces Are Easier to Fabricate and Inspect ► Stronger and Less Flaw Sensitivity
- Cost Effectiveness
  - Facility Upgrade Is Not Required
  - Shorter Schedule (Start With Design Plus Fab)
- Penalties Are Acceptable
  - Attachment Will Add Only 150 lbs

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COMPONENT: Quad Channel Fault Tolerant Controller (Three Channel Version Depicted)

FUNCTION: The Engine Controller is responsible for closed loop control of the NTRE engine and auxiliary power generation components.

The engine controller performs a complete engine system checkout and calibration prior to engine operation. This includes calibration of the individual instruments and control system components. During the start sequence the controller controls reactor reactivity, pump chill, turbopump ramp up, and power ramp up. The engine controller actively controls engine steady state operation to maximize engine Isp. Engine power down and post firing cool down is actively controlled to minimize propellant usage and maximize total engine Isp. The engine controller performs periodic engine system health monitoring and life prediction throughout engine operation.

Brayton cycle power generation is actively controlled throughout the mission. The controller is capable of adjusting the power output level over a 5:1 range to meet varying mission demands.

ARCHITECTURE: The NTRE engine controller is a 32 bit full voting four channel fault tolerant processor (FTP) that is derived from the Charles Stark Draper FTP architecture. The four channel controller provides full Fail Op/Fail Safe operation (higher levels of fault tolerance are available with degraded fault coverage).

Additionally the four channels are electrically and mechanically isolated from each other. This prevents a catastrophic electrical failure from propagating from one channel to the next.

The 32 bit Intel I80960 microprocessor provides the processing power for the engine controller. The I80960 is optimized to efficiently execute the Ada language. This central processor provides many advanced enhancements such as automatic exception handling and memory management that facilitate the efficient processing of Ada language. Over 2Mb of memory is provided on the digital computer unit module. This complement of memory is more than sufficient for both engine system control code and advanced health monitoring and life prediction algorithms.

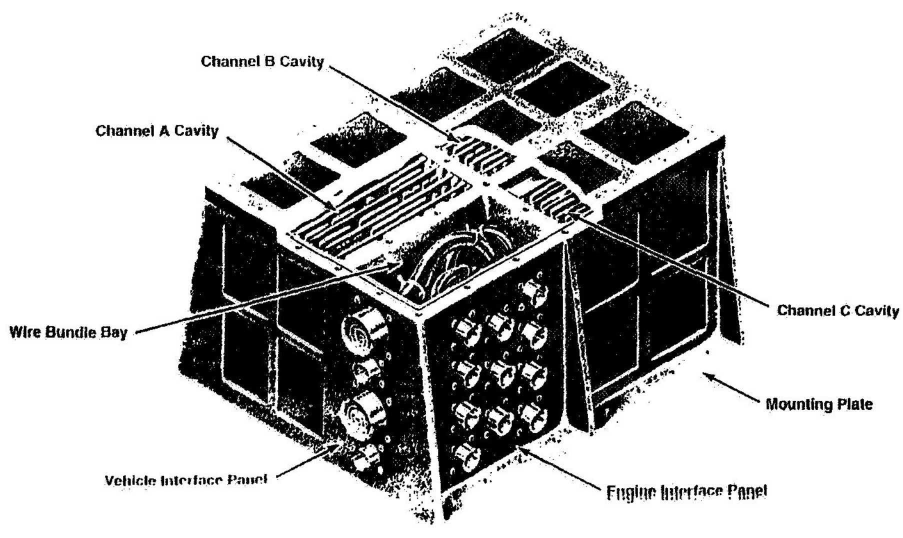
INTERFACES: 4 MIL-STD-1553B Vehicle Command Channels 4 MIL-STD-1553B Vehicle Data Channels, 4 MIL-STD-1553B Effector Command Channels, 4 Vehicle Power Buses, Solenoid Interfaces

SIZE: 8 in. x 16 in. x 10 in.

WEIGHT: 59 lbs (46 lbs Electronics + 13 lbs Shield)

TOTAL DOSE: 100 K RADs (SI)

Advanced Fault Tolerant Controller Improves Mission Reliability



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COMPONENT: Dual Channel Electro-Mechanical Actuator

FUNCTION: Provide actuation for modulating valves over a -1 to 90 deg arc. The EMA feature load insensitivity and high positional accuracy/repeatability.

ARCHITECTURE: The EMA is a fully redundant actuator featuring dual channel redundant bus interfaces, dual redundant power interfaces, dual channel redundant control electronics, dual electric redundant motors on a common shaft, and dual resolvers on a common shaft. The two EMA channels contain no electrical cross strapping and are mechanically isolated from each other. This prevents a catastrophic electronics failure in one channel from propagating to the other channel.

INTERFACES: 2 MIL-STD-1553B Valve Command Channels, 2 Power Buses

SIZE: 4 in. x 6 in. x 10 in.

WEIGHT: 31 lbs (9 lbs Electronics/Mechanics + 32 lbs Shield)

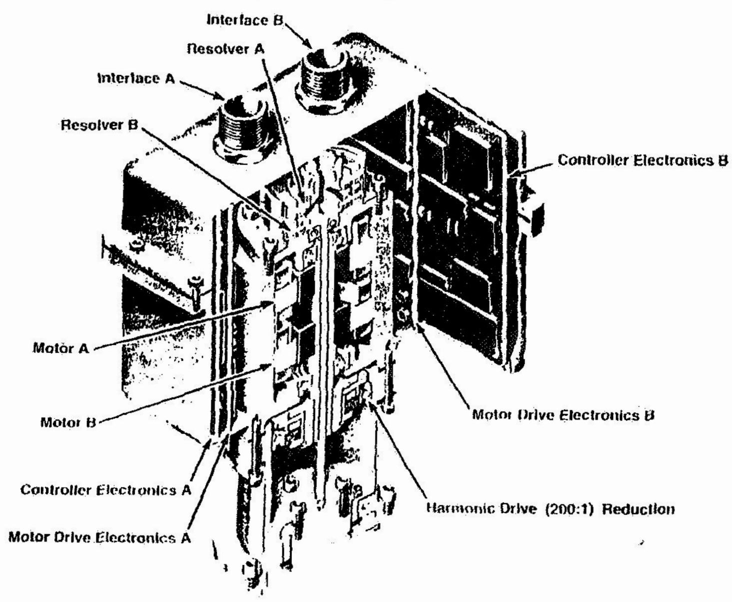
TOTAL DOSE: 100K RADs(Si)

TORQUE: 600 in.-lb

SLEW RATE: 360 deg/sec

POSITIONAL ACCURACY/REPEATABILITY: ± 0.5 deg

Advanced Electro-Mechanical Actuator Combines High Torque and Small Size



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COMPONENT: Quad Channel Control System

FUNCTION: Provide full Fail Op/Fail Safe engine and auxiliary power generation control.

ARCHITECTURE: The control system is designed with a high degree of symmetry and redundancy. The symmetry of the control system greatly reduces the complexity of the redundancy management software and improves system reliability and verifiability. Critical control valves such as the engine isolation valves are fully quadded thus allowing them to tolerate one stuck open or one stuck shut failure. Other valves are either simplex or dual (serial or parallel) depending upon the function of the valve.

There are two interfaces to the electro-mechanical actuators. These two interfaces are referred to as the Active Effector Control Bus and the Passive Effector Control Bus. Each control bus is actually a redundant 1553B implementation. This provides a total of four data paths to each actuator thus providing full Fail Op/Fail Safe capabilities of the control system.

Each solenoid actuator has dual coils. This provides fully redundant interfaces to the engine controller. Like the effector control buses the solenoid interfaces are organized as active and passive interfaces. Additionally solenoids contain a mechanical preload that forces the solenoid into a safe position in the event of a total interface failure.

Critical engine parameters, such as chamber pressure, are fully quad redundant. Other parameters such as moderator temperature are simplex or dual per moderator element as called for by FEMA/reliability analysis.

Heavy use will be made of sensor analytical redundancy techniques. These techniques allow a failed parameter to be substituted by using a physical model and related measurements.

INTERFACES: 2 MIL-STD-1553B Valve Command Channels, 2 Power Buses

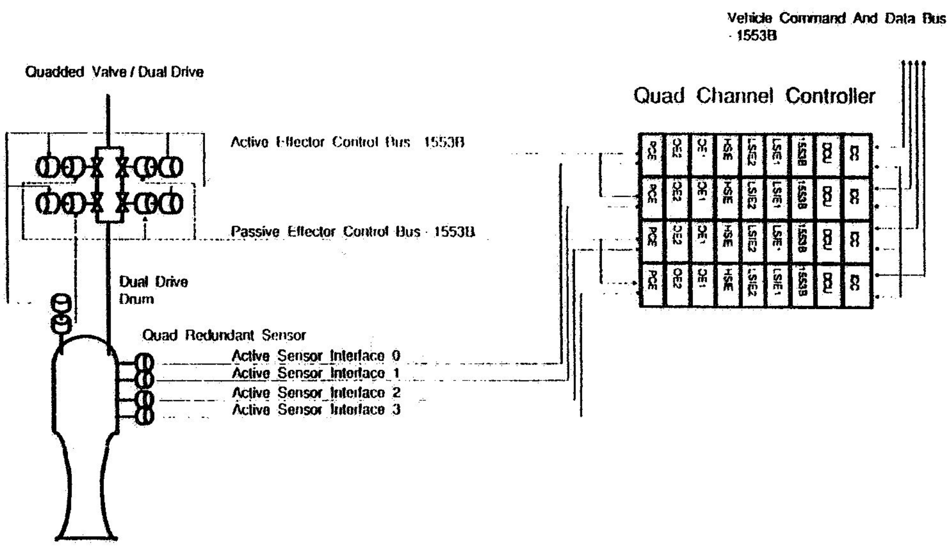
SIZE: 8 in. x 16 in. x 10 in.

WEIGHT: 31 lbs (9 lbs Electronics/Mechanics + 32 lbs Shield)

TOTAL DOSE: 100K RADs(Si)

TORQUE: 600 in.-lb

Quad Channel Control System Improves Mission Reliability



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COMPONENT: Operational Flight Program

FUNCTION: The Operational Flight Program (OFP) provides the control, health monitoring and life prediction capabilities seen in the control system. All of the dynamic engine control laws are found in the OFP. Engine system health monitoring and life prediction algorithms are also resident within the OFP. Additionally the OFP manages the interface hardware within the engine controller.

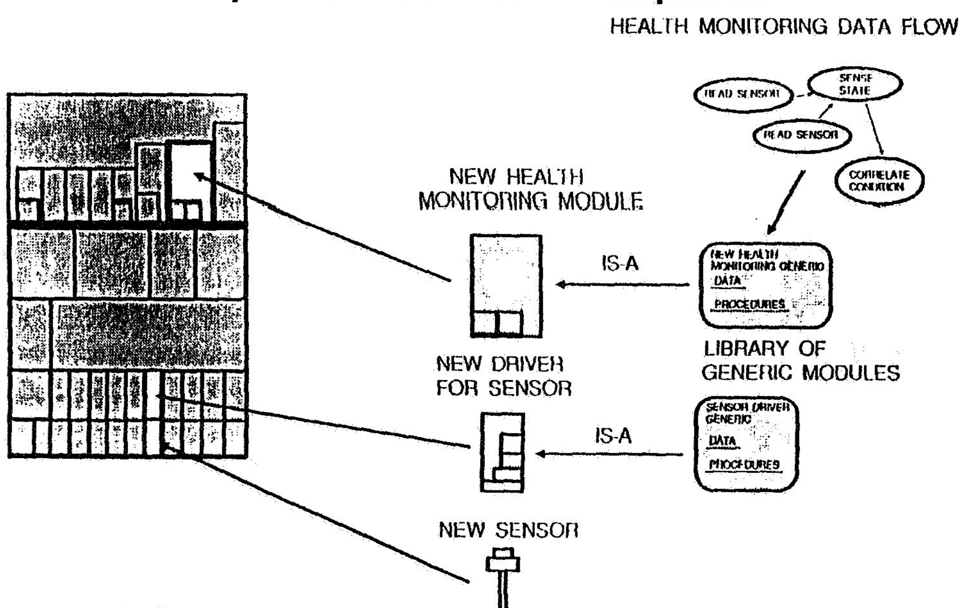
The OFP implements the required vehicle communications protocols and validates commands. Additionally status and data packets are sent back to the vehicle.

ARCHITECTURE: The OFP design is based on a highly modular, structured, functional decomposition of the required functionality. Related functions are combined into modules. Thus all the engine control functions are grouped into the Engine Control Module; all the Health Monitoring functions are grouped within the Health Monitoring Module. Modules have rigid functional, interface, protocol, and temporal specifications. These specifications minimize the interactions between modules, increasing software reliability and reducing verification and validation efforts.

The modular architecture allows individual modules to be upgraded throughout the life of the NTRE program while preserving the software investment. Modules are designed to be "plugged in" in a manner similar to mechanical components thus reducing the costs associated with software verification and validation.

INTERFACES: Controller/IO Devices

Plug In Software Modules Improves Controller Development



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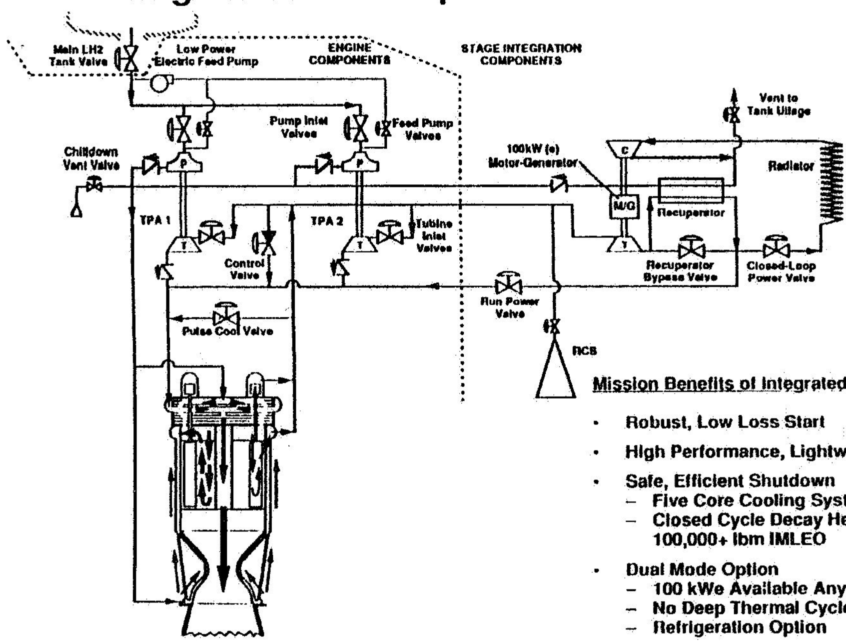
Integrated Engine

Mel Bulman

Integrated NTRE Improves Mission Performance

The SEI stage will require many subsystems in addition to the engine and tanks. Our Integrated NTRE includes a number of systems normally assigned to the stage. It can provide reaction control and orbit maneuvering thrust during coast. During the main burn, the engine can provide autogenous tank pressurization and electrical power. After shutdown, the reactor can be used as a heat source for generating up to 100 kW (e) per engine. All of this is accomplished at lower weight than if separate systems are employed to achieve these functions.

Integrated NTRE Improves Mission Performance



Mission Benefits of Integrated NTRE

- Robust, Low Loss Start
- High Performance, Lightweight Engine
- Safe, Efficient Shutdown
  - Five Core Cooling Systems
  - Closed Cycle Decay Heat Removal Saves 100,000+ lbm IMLEO
- Dual Mode Option
  - 100 kWe Available Any Time During Mission
  - No Deep Thermal Cycles
  - Refrigeration Option
- OMS and RCS Impulse Available at High Isp
- One or Two Cold Flow Starts/Mission

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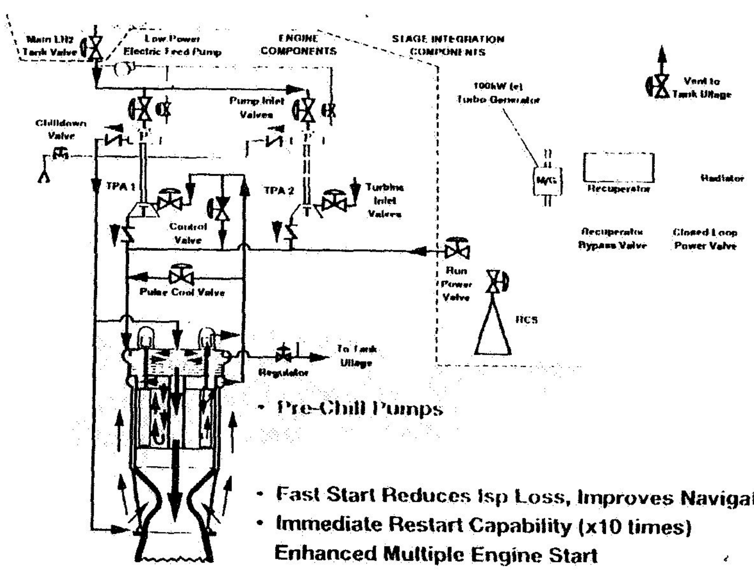
Integrated NTRE Start Sequence

- Engine Prestart Conditioning
  - Pump Chill In
  - Moderator Loop Pressurization With TPA Chill H_2 (First Start Only)
  - Closed Loop Engine Warm Up (First Start Only)
  - Engine Now on Standby Mode For Starting
- Start
  - Spin Start TPAs With Warm Pressurized H_2 From Moderator Loop
  - TPA Acceleration Dominated by Engine Thermal Mass (Power For Approximately 10 Starts in Recuperator Alone)

Integrated Engine Increases Start Reliability and Safety



- Pre-Chill Pumps
- Fast Start Reduces Isp Loss, Improves Navigational Accuracy
- Immediate Restart Capability (x10 times)
  Enhanced Multiple Engine Start

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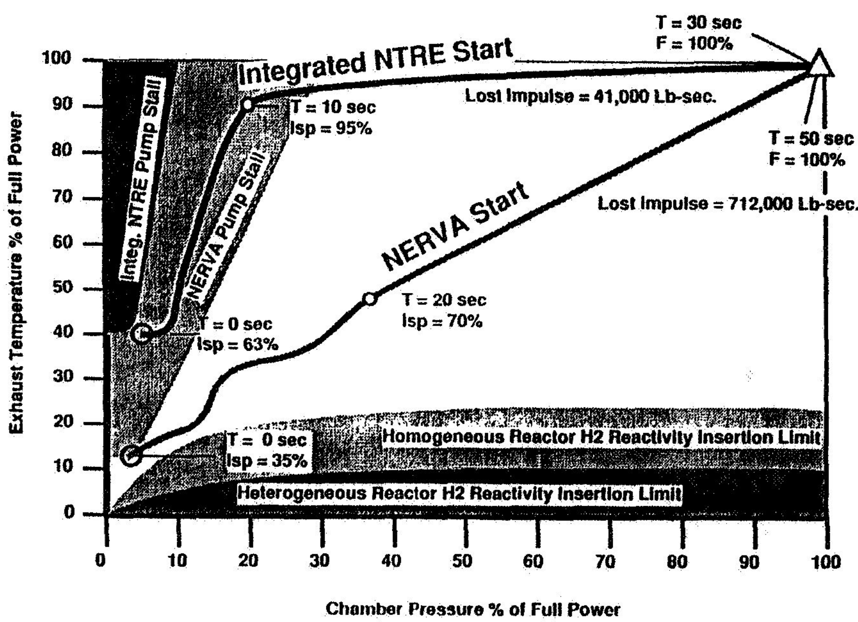
ORIGINAL PAGE IS OF POOR QUALITY

Our Propellant Feed System Dynamics Are Efficiently Controlled

- Engine Prestart Conditioning
  - Pumps Chilled In
  - Reactor Warmed
  - Feed System Pressurized (Reduces Inrush Dynamics)
- Aerojet Pumps Are Designed With Greater Stall Margin
- Our Recuperated Cycle Greatly Aids the Start Up
  - Ample Thermal Power Accelerates Bootstrap
  - Provides Thermal and Hydraulic Damping
  - Isolates Fuel Assembly From Feed System
- Our Integrated Controller Can Choose the Optimum Path to Full Power, Balancing:
  - Isp Loss
  - Fuel Element Thermal Shock

Our Integrated Engine Starts More Reliably

And With Less Impulse Loss than Nerva Type Engines



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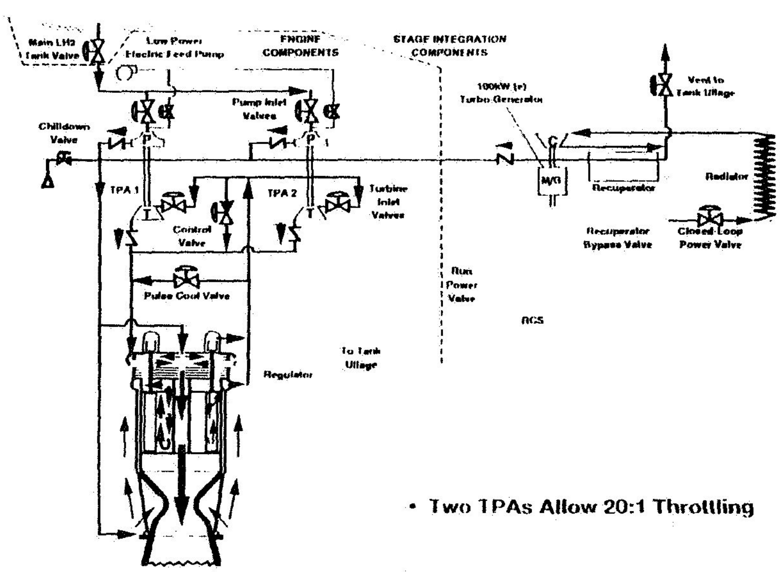
Tapping into the hot H_2 in the moderator loop of our integrated NTRE during operation allows us to generate up to 100 kW(e), attitude control impulse, and tank pressurization at lower cost than if provided by separate systems.

Integrated Engine Provides RCS and Tank Pressurization During "Burns"



- Two TPAs Allow 20:1 Throttling

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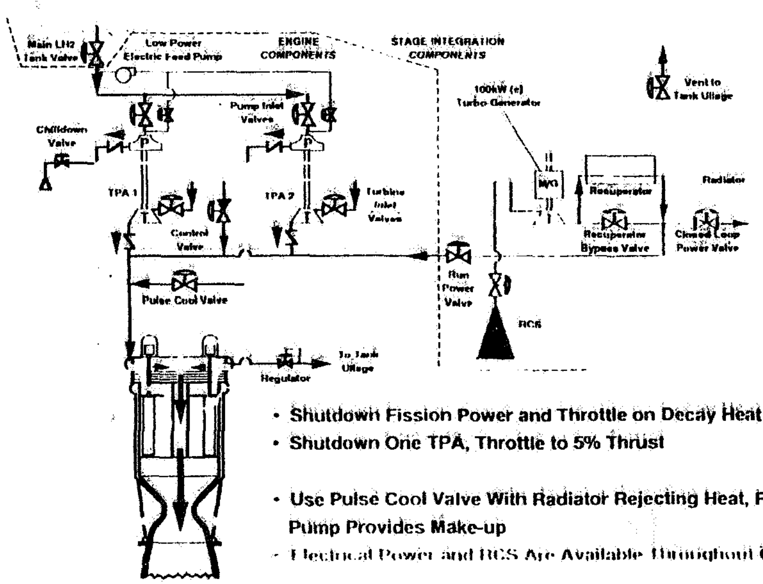
The decay heat build up in the engine during the main burn must be removed from the reactor or it will over heat. Decay heat persists for days even after a short fifteen minute burn. Our integrated engine can reject a significant portion of the heat through its radiators, greatly reducing the expenditure of propellant to cool the engine. This reduces vehicle mass (IMLEO).

Integrated Engine Saves Over 100,000 lbm LH_2



- Shutdown Fission Power and Throttle on Decay Heat
- Shutdown One TPA, Throttle to 5% Thrust
- Use Pulse Cool Valve With Radiator Rejecting Heat, Feed Pump Provides Make-up
- Electrical Power and RCS Are Available Throughout Cooldown

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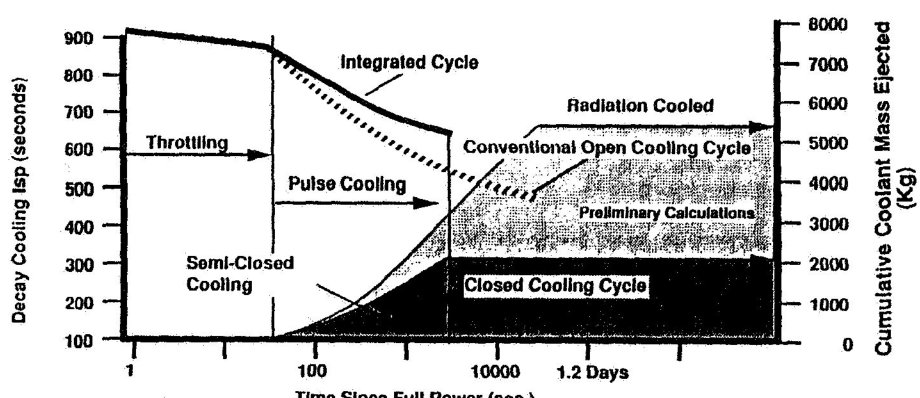
Our Closed Cycle Cooling System Saves Over 7500 lbm During Perigee Pulsing

Conventional Cooling System	Pluse 1	Cool 1	Pluse 2	Cool 2	Pluse 3	Cool 3
Initial Mass (Lbm)	1000000	850842	840942	695357	684665	542970
Burn Time (Sec)	981	174 Min	958	1006 Min	932	Days
Propellant Consumed (Lbm)	149158	9900	145586	10692	141694	10692
Effective Isp (Sec)	915	640	915	600	915	600
Final Mass (Lbm)	850842	840942	695357	684665	542970	532278
ΔV (ft/sec.)	4755	241	5596	299	6825	384
Mission Velocity (Ft/Sec.)	4755	4996	10591	10890	17716	18100
Mission Effective Isp (Sec)		896	906	894	902	892
Closed Cycle Cooling System						
	Pluse 1	Cool 1	Pluse 2	Cool 2	Pluse 3	Cool 3
Initial Mass (Lbm)	1000000	845720	843520	692280	690080	541986
Burn Time (Sec)	1015	174 Min	995	1006 Min	974	Days
Propellant Consumed (Lbm)	154280	2200	151240	2200	148094	2200
Effective Isp (Sec)	915	760	915	760	915	760
Final Mass (Lbm)	845720	843520	692280	690080	541986	539786
ΔV (ft/sec.)	4932	64	5816	78	7111	99
Mission Velocity (Ft/Sec.)	4932	4996	10812	10890	18001	18100
Mission Effective Isp (Sec)		913	914	913	914	913
Propellant savings (lbm)		2578		5415		7508

Mission Average Isp Improved 21 Seconds With Closed Cycle Cooling

Closed Cycle Cooling Reduces IMLEO Over 8%

950 Sec. Burn @ 150,000 lbf (2-3 Engines)



Closed Cycle Benefits

- Reduced Coolant Ejection
- Higher Ejection Isp (cut off low Temp. Tail)
- 75+%Reduction Mission Isp Loss f(ΔIsp & ΔMass)
- Reduced g Loss (Cooldown Impulse Delivered Faster)
- Closed Cycle Has All Brayton Power Cycle Components (Except Generator)

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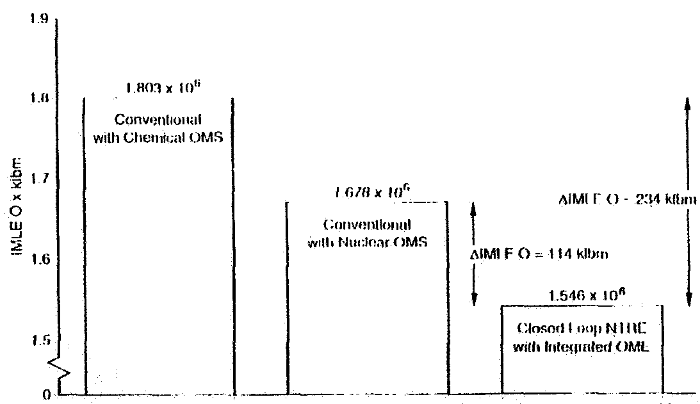
MJB 10/1592

Our Integrated NTRE Can Reduce IMLEO 100-200+ KLbm

Conventional Cooling System (H2/O2 OME System)	Burn 1	OMS-1	Burn 2	Burn 3	OMS-2	Burn 4
Initial Mass (Lbm)	1803170	871599	813424	397457	243069	226845
Propellant Consumed (Lbm)	846882	58175	287242	140353	16224	106541
Effective Isp (Sec)	892	450	892	892	450	892
Final Mass (Lbm)	956288	813424	526182	257104	226845	120304
ΔV (ft/sec.)	18200	1000	12500	12500	1000	18200
Mission Velocity (Ft/Sec.)	18200	19200	31700	44200	45200	63400
Δ IMLEO (Klbm/%) Over Integ. NTRE	257	16.64		Payload Returned		50000
Conventional Cooling System With Main Engine Restart for Plane Change Maneuver	**Burn 1**	**OMS-1**	**Burn 2**	**Burn 3**	**OMS-2**	**Burn 4**
Initial Mass (Lbm)	1678020	811106	775875	374494	229025	219077
Propellant Consumed (Lbm)	788104	35231	273983	132244	9948	102893
Effective Isp (Sec)	892	700	892	892	700	892
Final Mass (Lbm)	889916	775875	501892	242250	219077	116185
ΔV (ft/sec.)	18200	1000	12500	12500	1000	18200
Mission Velocity (Ft/Sec.)	18200	19200	31700	44200	45200	63400
Δ IMLEO (Klbm/%) Over Integ. NTRE	132	8.55		Payload Returned		50000
Integrated NTRE Closed Cycle Cooling System + Nuclear OMS (without restart)	**Burn 1**	**OMS-1**	**Burn 2**	**Burn 3**	**OMS-2**	**Burn 4**
Initial Mass (Lbm)	1545880	760485	727453	350094	216612	207203
Propellant Consumed (Lbm)	713995	33032	252145	121347	9409	95701
Effective Isp (Sec)	913	700	913	913	700	913
Final Mass (Lbm)	831885	727453	475308	228746	207203	111502
ΔV (ft/sec.)	18200	1000	12500	12500	1000	18200
Mission Velocity (Ft/Sec.)	18200	19200	31700	44200	45200	63400
				Payload Returned		50000
				100 Klbm Drop @ Mars, 10% Tank fraction		
				Tanks droped after burns		

10/18/92

Closed Cycle Cooling Can Reduce IMLEO Over 100K lb*



* Four Burn Full Up Mars Mission

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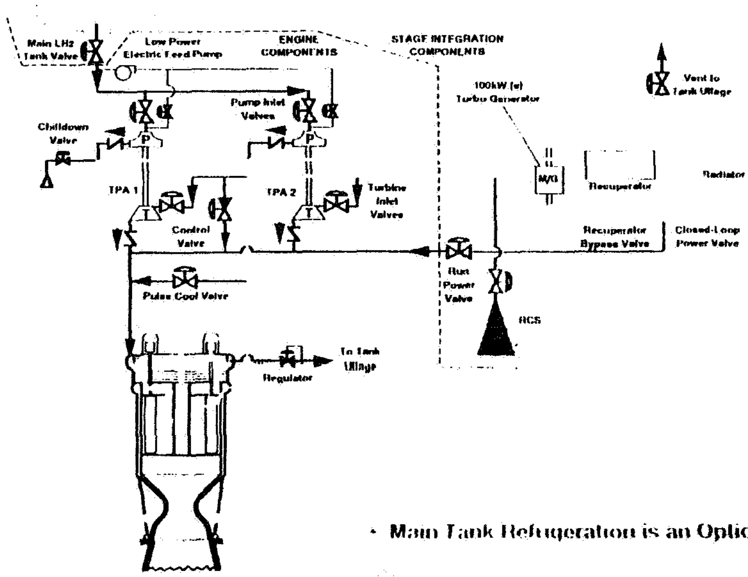
During coast, our integrated engine is kept warm while generating up to 100 kW(e). The mean electrical power will be less than 100 kW(e). At 20 kW(e) the Brayton cycle efficiency is approximately 30% requiring a thermal power of approximately 60 kW(t), which causes only a small additional burn up of the reactor fuel.

Integrated Engine Eliminates Additional Stage Components and Improves Stage Performance



• Main Tank Refrigeration is an Option

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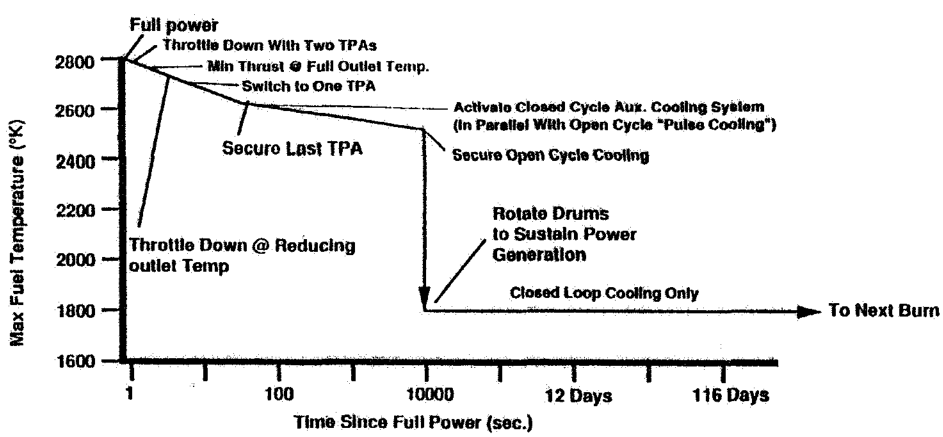
The integrated engine adds life to the fuel, because it allows fuel to be kept warm during coast (~1800 K). Therefore, the fuel will see only ~1000K ΔT's during startup and shutdown. 1800K was chosen to balance the effects of vaporization rate, thermal cycling, and power cycle efficiency.

Engine Does Not Cool Fully Between Major Burns



Aerojet Cycle Benefits
- Reduced Thermal Shock
- Integrated Power Supply
- ACS and OMS Power
- Full Thrust Available on Short Notice

Reliability and Safety

Mel Bulman

GenCorp Aerojet • Energopool • Babcock & Wilcox

Our Turbopump System Improves Mission Reliability*

- Single TPA System Has ~ 4 Times the Probability of Total Failure vs 2 TPAs
- Twin Spool 4 Stage Pump Is More Reliable Than Single Shaft TPAs at the Same Discharge Pressure

* Industry Standard Component Failure Rates Applied to Feed Systems

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Dual Spool TPA Provides High Margin for NTRE

- Impellers Stressed Less for Same Weight and Performance – Less ΔP Per Impeller
- Four Bearings to Share Loads Rather Than Two
- Unshrouded, Machined Impellers Have Higher Strength Than Casting
- Runs Subcritical for Less Dynamic Stress

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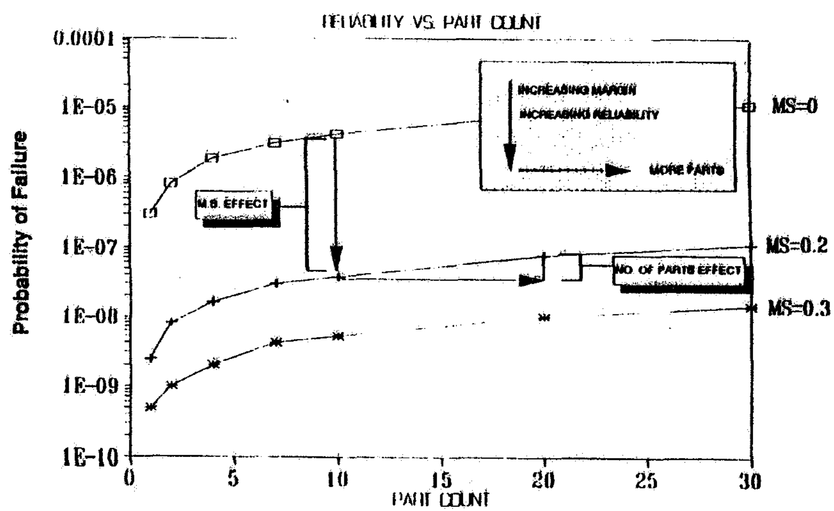
Reliability Increased With Lower Stressed Parts



Margin A Stronger Influence On Reliability Than Parts Count

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NTRE – Concept Design

Failure Modes and Effects Analysis

A Failure Modes and Effects Analysis was completed for the Particle Bed Reactor concept. The failure modes of the major components were identified. The criticality and effect of each mode was determined and possible ways to minimize the occurrence of each mode were also identified. This analysis will be expanded and updated during the preliminary design phase to reflect design maturation. The FMEA will be the basis for developing a Reliability Fault Tree, which will show system interactions graphically. Quantitative evaluation of the base events of the Fault Tree will allow reliability predictions to be made of the system. The Fault Tree will be developed using CAFTA, a Computer Aided Fault Tree Analysis code, which facilitates reliability and system safety analysis of complex systems.

NTRE – Concept Design

Failure Modes and Effects Analysis

- Preliminary FMEA Has Been Completed for Engine Concept
  - Component Failure Modes Identified
  - Actions to Minimize Occurrence Are Incorporated
    - Redundancy
    - Robust Design
    - Adequate Testing and Inspection
- FMEA to Be Updated During Design Phase to Reflect Maturing Design

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NTRE – Concept Design

Reliability

Methodology to Evaluate the Effect of Redundant Components on the System

Reliability block diagrams and industry standard failure rates of like components will be used to assist in the evaluation of the effect of redundancies of various components on the reliability of the system.

A system level Fault Tree will be developed which will be used to analyze the reliability of the system during the various operating phases of the proposed mission.

A system Fault Tree has the advantage of being able to see graphically the interactions of a complex system. It is difficult to model these interactions using only block diagrams. Block diagrams are useful in studying effects on the system of series redundancy or parallel redundancy of a few components. But the overall system reliability is better evaluated by doing a quantitative evaluation of a system Fault Tree.

A reliability Fault Tree differs from a system safety Fault Tree only in the definition of the top event. Process and human errors resulting in system failure are always included in the system safety Fault Tree. They can also be included in the Reliability Fault Tree. If the purpose of the Reliability Fault Tree is to assess the reliability of the design then the possible process and human errors would not be included in this Fault Tree.

NTRE – Concept Design

Reliability

Methodology to Evaluate the Effect of Redundant Components on the System

- Reliability Block Diagrams and Failure Probabilities Using Industry Standard Failure Rates for Like Components Will Be Used to Evaluate Need for Redundant Components
- Reliability Fault Tree Will be Developed of Overall System to Be Used in Evaluating System Reliability

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NTRE Design Criteria

- Safety Factors

  Pressure Loads

Yield Safety Factors	FSy	=	1.25
Ultimate Safety Factor	FSu	=	1.50

  Thermal Loads

Yield Safety Factor	FSy	=	1.00
Ultimate Safety Factor	FSu	=	1.00

- Margin of Safety

  MS = (Allowable Stress) (FS * Applied Stress) – 1.0

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17/80 5bio/38

System Safety

Design Includes Hazard Elimination and Control Provisions for Wide Range of Potential Hazards

- A Preliminary Hazard Analysis Has Been Accomplished
- This Analysis Is the Initial System Safety Task Which Included a Review of the NTRE Components, Potential Hazardous Conditions, Effect on the System if an Undesirable Condition Took Place, and Recommended Controls in the Design to Prevent an Occurrence From Happening
  - In Addition to Component Review, Natural Environment, Oxygen Rich Environment, and Aerospace Ground Equipment Hazards Were Considered
- This Study Illustrates a Combination of 17 Hazardous Conditions That Were Considered

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Nuclear Safety

- Water Immersion
  - Most Reactor Voids Filled With Poison During Launch
  - Reactor Design Remains Subcritical
    - Fuel Internally Retained
  - Launch Safety/Emergency Shutdown Procedures
  - Qualification and Acceptance Tests
- Impact Compaction (Ground)
  - Collision Reduces Void Fraction
  - Reactor Design Ensures/Remains Subcritical
    - Fuel Internally Retained
  - Launch Safety/Emergency Shutdown Procedures
  - Qualification and Acceptance Tests

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Nuclear Safety
Emergency Cooldown Procedure

- 1 TPA Failure
  - Fast (~ 1 sec) or Slow (~ 30 Seconds) Single Failure
    - Normal Reactor Shutdown
      Cooldown at High Power With 2nd TPA
    - Continue Mission With 2nd TPA
- 2 TPA Failure
  - 1 Fails Fast, 1 Fails Slowly
    - Shutdown Reaction at High Power With Slowly Failing TPA
    - Employ Pulsed Cooling System Prior to 2nd TPA Failure
- 2 TPA Failure
  - 2 Fail Fast or Nearly Simultaneously
    - Shutdown Reactor
    - Cooldown at High Power With Crossover System

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Nuclear Safety

Reactor Leakage Potential Is Minimized

- Use of Non-Radiation Embrittlement Materials
- Maintain Within Temperature Extremes
- Develop Approved Installation Procedures
- Post-Reactor Installation Leak Checks
- Test Area Monitored for Leakage
- Non-Nuclear Qualification and Acceptance Tests

Minimize Leakage Effect

- Radiation Hardened Material/Electronics
- Shielding
- Qualification and Acceptance Tests

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17 69 5b/i/43

Nuclear Safety

Design Controls Radiation Hazards to a Minimum

- Shielding Material – "Burn-In" Process
- Internal Shielding
  - Attenuates Levels at Propellant Tank
  - Reduces Levels to Engine and Stage Components
- External Shielding
  - Reduces Level to Crew and Stage
- Components Nuclear Hardened
- Proof-of-Concept Testing

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17 69 5b/i/44

Nuclear Safety

Reactor Risks and Hazards Are Minimized, i.e., System Design

- Controller Architecture
  - Diagnostic Instrumentation Monitors Reactor
  - Quad Channel Fault Tolerance Operation
  - Software Redundancy Design
  - Multiple Signals Required to Activate Valves
- Reactor
  - Control Drums Have Redundant Drive Motors and Couplings
  - Safety Rods Have Redundant Drive Motors and Couplings
- Shielding of Safety Rod Drives Ensures Rod In or Out Control Capability
- Non-Nuclear Vibration, Thermal and Shock Tests to Verify Structural Integrity

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17 60 5mu45

Nuclear Safety

Design Controls All Identified Energy Sources

- Reactor
- Pressurized Propellant Feed Lines/Fittings/Valves
- Turbopump Assembly
- Pyrotechnic Isolation Valves
- Electronics
- Hydrogen in Tank

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17 69 5mu46

System Safety

Summary

- Nuclear Safety Hazards Will Be Controlled Through Preventative Measures
  - Margins
  - Redundancy
  - Diversity
- NTRE Design Will Meet the Applicable Safety Requirements for Operation on the Eastern Test Range or Western Test Range
- The APD/B&W Team Will:
  - Ensure Design Meets Proof-of-Concept Objectives With Risk Reduced to as Low as Reasonably Achievable
  - Support the Nuclear Safety Policy Working Group Recommendations as Applicable

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PBR Engine Sensitivity Studies

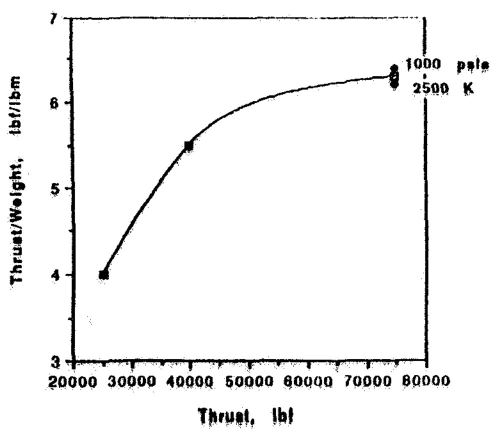
Mel Bulman

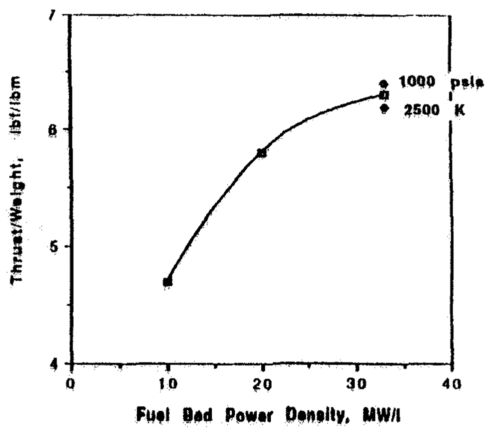
GenCorp Aerojet • Energopool • Babcock & Wilcox

	CIS*	PBR	40k PBR	25k PBR	2500K PBR	10MW/l	20MW/l	1000 psia
Thrust, lbf	75000	75000	40000	25000	75000	75000	75000	75000
Chamber Pressure, psia	2000	2000	2000	2000	2000	2000	2000	1000
Nozzle Area Ratio, Ae/At	300	300	300	300	300	300	300	200
Engine Specific Impulse, sec	959	915	915	915	880	915	915	910
Mars Mission Specific Impulse, sec	930	887	887	887	853	887	887	883
Engine Total Weight, lbm	15900	11879	7278	6193	12027	15901	13032	11687
Thrust/Weight	4.7	6.3	5.5	4.0	6.2	4.7	5.8	6.4
Engines per Vehicle	2	2	4	6	2	2	2	2
Payload Returned to Earth, lbm	47067	44900	37947	26897	34682	34242	42201	44222
Engine Weight Breakdown								
Component	Weight, lbm							
Uncooled Nozzle	232	240	175	139	249	240	240	212
Cooled Nozzle	965	1000	730	577	1039	1000	1000	1414
Pressure Vessel, Reactor Manifolds & CSS	2536	1501	701	613	1503	2168	1694	1042
Reactor, Reactor I&C	6613	3982	2244	2019	3982	8219	4603	3957
Turbopump Assemblies (2)	410	410	236	152	440	410	410	300
Recuperator / Shield	2425	2168	1111	727	2179	2914	2395	2412
Secondary Shield	642	521	572	753	523	704	576	565
Plumbing/Valves	1320	1320	964	762	1372	1320	1320	1048
Controls and Shielding	758	737	544	451	740	926	794	737
NTRE w/ Stage Power/Heat Removal								
Stage Power & Heat Removal Sys Wt, lbm	2000	2000	1285	939	2000	2000	2000	2000
Engine with Power Sys Wt, lbm	17900	13879	8563	7132	14027	17901	15032	13687
Mars Mission Specific Impulse, sec	949	905	905	905	871	905	905	901
Payload Returned to Earth, lbm	52929	50226	43249	32161	39768	39531	47518	49523

* CIS Engine Fuel Bundle Power Density is 12 MW/liter. Therefore we expect its weight to scale approximately as the PBR engine with power density.

High Thrust and Power Density Increase Engine Thrust-to-Weight





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PBR Design

Richard Rochow

Our PBR engine concept is best summarized by including the rationale behind the selection of each major subsystem concept or operating parameter. 2700 K mixed mean reactor outlet gas temperature is selected by B&W fuel experts to meet the 4.5 hour life requirement with an appropriate margin by the end of fuel assembly development. The engine design/operating selection of 300:1 nozzle area ratio and nozzle inlet pressure of 2,000 psia is the result of a Mars mission payload trade study; it gives the best combination of engine specific impulse and weight. A recuperated turbopump drive cycle was selected for several reasons: (1) nozzle inlet pressures of 1000 psia and above are enabled by recycling topping heat through the turbine, and no reactor manifolding need be added to extract turbine drive heat directly from the core, (2) engine start and shutdown transients are smoothed and assisted by the large, available heat capacity of the high surface area recuperator, (3) the steel heat exchanger adds no weight to the engine, because its weight is determined by its other duties as a large part of its internal gamma shield and as the forward closure of the reactor vessel. A forward core support structure was selected, largely because it is the American experience base. The forward structure is cool, it forms part of the gamma shield, and it is used as propellant manifolding. Fuel assemblies operate in tension and are constructed of steel and beryllium, according to U.S. experience. A single DeLaval nozzle is selected that is internally cooled with hydrogen; no hydrogen bleed is necessary, because our formed platelet nozzle operates with low internal wall temperatures at high heat fluxes. A 40 Klbf nozzle of similar design, material and coolant is now in test at NASA. The nozzle is small, because of the engine's high operating pressure, and we use a low weight, carbon-carbon nozzle extension. Its surface may be converted to ZrC to improve its life in hydrogen environment, using near term technology processes similar to those in work at Aerojet, however this is probably unnecessary, because total surface recession in 4.5 hours of operation is expected to be less than 0.025 in. (0.64 mm). A Li_6H neutron shield is encapsulated in aluminum and located external to the recuperator in order to simplify the core support structure. A secondary gamma shield is located at its forward face to provide sufficient gamma attenuation at all times during engine life. Both shields are cooled by propellant flow.

Design Rationale for NTRE With PBR Is Clear

	Design Parameter	Rationale
1	H_2 Mixed Mean Outlet Temperature (2,700K)	Expected 4.5 Hours Fuel Life
2	Pc (2,000 psia)	Best Mars Mission Performance With Reusable Engines
3	Ae/At (300)	
4	Power Cycle (Recuperated)	• Enable High Pressure With Simple Reactor • Enhances Transient Operation • No Weight Penalty (γ Shield)
5	Core Support (forward)	U.S. Experience Base
6	Nozzle (Single DeLaval) (H_2 Cooled Without Bleed Flow) (C-C Extension)	Near Term Technology Use (Formed Cu Platelet)
7	Neutron Shield (External)	Simplifies Core Support

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AEROJET/B&W PBR NTRE

NASA LeRC Final Report

Prepared by R.F. Rochow
Babcock & Wilcox, ASE
Oct 23, 1992

REACTOR DESIGN PHILOSOPHY

BABCOCK & WILCOX HAS APPLIED ITS KNOWLEDGE OF THE PARTICLE BED REACTOR (PBR) TO MEET THE NASA DESIGN REQUIREMENTS. THE OBJECTIVES OF THE DESIGN WERE TO STAY WITHIN THE TECHNOLOGY BASE FOR THE PBR. THIS INCLUDES THE MIXED MEAN EXHAUST GAS TEMPERATURE, ENGINE PERFORMANCE AND PRIMARILY THE SYSTEM SAFETY. THE PBR IS CAPABLE OF VERY HIGH T/W RATIOS HOWEVER, FOR MAN-RATED SYSTEMS OUR BASELINE DESIGN HAS BEEN CONSERVATIVELY DESIGNED AND INCORPORATES ROBUST AND THEREFORE RELIABLE COMPONENTS. THE PBR CONCEPT HAS THEREFORE INCURRED SOME MASS PENALTIES WHICH ARE BELIEVED TO BE PRUDENT IN TERMS OF SAFETY FOR THE CREW.

REACTOR DESIGN PHILOSOPHY:

STAY WITHIN THE TECHNOLOGY

- NASA Requirements Isp/Gas Temperature
- Mass Penalties Accepted
- Highest Possible Performance is *NOT* Top Priority

- *SAFETY IS*-

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PBR PROVIDES A COMPACT HIGH POWER ENGINE

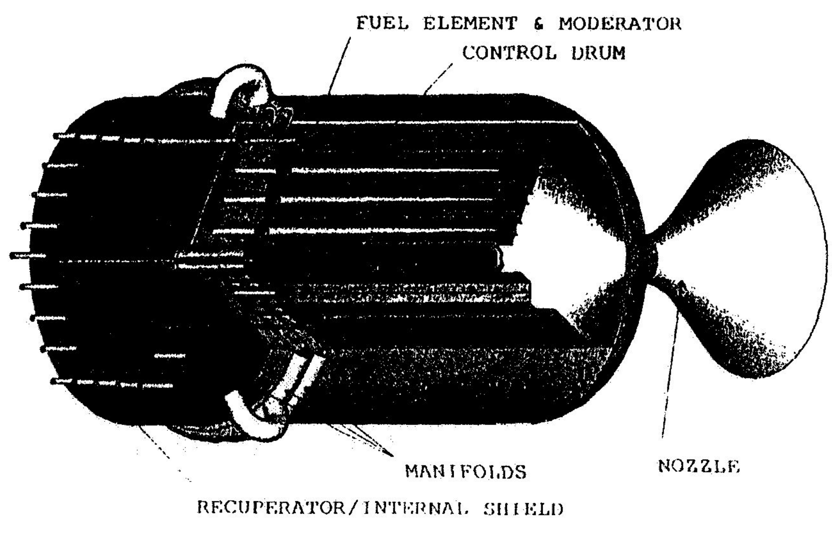
THE PBR IS A PARTICULARLY ATTRACTIVE CONCEPT DUE TO THE HIGH SURFACE AREA OF THE PARTICLE FUEL FORM. THE HEAT TRANSFER CAPABILITY IS UNSURPASSED. SMALL PARTICLES, BY DEFINITION, ALSO LIMIT THE THERMAL STRESSES WITHIN THE PARTICLES DUE TO THE SHORT CONDUCTION PATH LENGTH. THESE FEATURES ALLOW THE PBR TO OPERATE AT HIGH POWER DENSITY AND THEREFORE REQUIRE LESS CORE VOLUME. SMALLER CORE VOLUME TRANSLATES TO REDUCTIONS IN VESSEL DIAMETER AND SHIELD DIAMETER. ALL OF WHICH TRANSLATES TO LOWER MASS (OR HIGHER T/W).

THE HETEROGENEOUS CORE UTILIZES 36 FUEL ELEMENTS ON A PITCH OF 11 CM. THE SUPPORT FOR THE CORE IS ACCOMMODATED BY THE MANIFOLD STRUCTURE. THE CORE IS "HUNG" FROM THE TOP WHERE COOL GAS KEEPS THE STRUCTURE WITHIN ITS ALLOWABLE TEMPERATURE REGIME. BERYLLIUM MODERATOR HEXAGONS SURROUND EACH FUEL ELEMENT AND ACT AS ITS PRIMARY STRUCTURAL SUPPORT.

A TOTAL OF 18 ROTATING CONTROL DRUMS SURROUND THE CORE. THESE DRUMS CONTAIN A NEUTRON POISON (B_4C) SEGMENT TO CONTROL THE POWER OF THE REACTOR. A SAFETY ROD IS LOCATED AT THE CENTERLINE OF THE CORE. IT IS AN AXIALLY CONTROLLED POISON ROD (B_4C) AND IS A REDUNDANT SHUT-DOWN SYSTEM IN THE EVENT THE CONTROL DRUMS BECOME INOPERABLE.

THE RECUPERATOR SERVES MANY FUNCTIONS AND IS NECESSARY FOR POWER BALANCE. IT "RECYCLES" WASTE THERMAL ENERGY TO DRIVE THE TURBOPUMPS. IN ADDITION, IT PROVIDES A SIGNIFICANT PORTION OF THE INTERNAL SHIELDING FOR THE TURBOPUMPS. PLATELET TECHNOLOGY WILL BE USED TO FABRICATE THE RECUPERATOR AND THE COOLED PORTION OF THE NOZZLE.

PBR Provides a Compact High Power Engine



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ORIGINAL PAGE IS OF POOR QUALITY

RECUPERATED CYCLE PROVIDES SUPERIOR ENGINE OPERATION

THE RECUPERATED CYCLE BENEFITS THE ENGINE IN MANY WAYS. THE RECUPERATOR CONTRIBUTES TO THE SHIELDING REQUIREMENTS (PARTICULARLY GAMMA). IT ENABLES HIGH CHAMBER PRESSURE BY SATISFYING THE POWER BALANCE OF THE SYSTEM. IT ALSO HOLDS A LARGE AMOUNT OF THERMAL ENERGY. THERE IS SUFFICIENT THERMAL ENERGY WITHIN THE RECUPERATOR FOR SEVERAL RESTARTS. FROM THE REACTOR STANDPOINT, PERHAPS THE MOST IMPORTANT BENEFIT OF THE RECUPERATED CYCLE IS THAT IT PREVENTS LIQUID HYDROGEN FROM ENTERING THE CORE. THIS ELIMINATES THE POSSIBILITY OF HIGH EXCESSIVE REACTIVITY DUE TO VERY DENSE HYDROGEN IN THE CORE. IN ADDITION, THE RECUPERATOR DAMPS HYDRAULIC OSCILLATIONS THROUGH THE USE OF SMALL PASSAGES AND FLOW DIRECTION CHANGES. IT INSURES THE DELIVERY OF UNIFORM, STEADY FLOW TO THE FUEL ELEMENTS WITHOUT SHARP PRESSURE PULSES.

Recuperated Cycle Provides Superior Engine Operation

- Provides Cooled, Internal Gamma Shield
- Enables High Chamber Pressure
- Provides Thermal Energy for Turbopump Start
  - Energy available for many starts
- **Provides Safe, Controllable Reactor Start**
  - **Prevents Liquid Hydrogen entry into the core**
  - **Decouples and damps system oscillations**

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REACTOR SUMMARY: KEY SPECS

THERE ARE SEVERAL NOTEWORTHY REACTOR SPECIFICATIONS. PERHAPS THE MOST IMPORTANT IS THE AVERAGE POWER DENSITY THAT WAS CHOSEN FOR THE BASELINE. THE VALUE OF 33 MW/L WAS CHOSEN FOR THIS MISSION BECAUSE WE BELIEVE IT IS ACHIEVABLE. FURTHERMORE, THE RUSSIAN ENGINEERS HAVE DEMONSTRATED UP TO 90 MW/L (FOR MINUTES) WITH SIMILAR FUEL COMPOSITION. IT IS IMPORTANT TO NOTE THAT THE POWER DENSITY DETERMINES THE SIZE OF THE REACTOR AND THEREFORE THE ENGINE SIZE. THE CHANGE IN MASS OF THE REACTOR WITH POWER DENSITY VARIATIONS IS SHOWN SEPARATELY.

THE BASELINE FUEL COMPOSITION IS (U,Zr)C WHICH IS COATED WITH NbC. THIS COMBINATION OFFERS HIGH TEMPERATURE CAPABILITY (IE. MELT IS APPROX. 3,300-3,400K) AND THE COATING PROVIDES THE FISSION PRODUCT RETENTION AND IS SELECTED TO MATCH THE THERMAL EXPANSION OF THE BINARY FUEL BETTER THAN ZrC.

REACTOR SUMMARY: *KEY SPECS*

○ Reactor Power	1.56 GW
○ Thrust (200:1 nozzle)	75,000 Lb
○ Gas Outlet Temp (mixed mean)	2,700 K (4,400F)
○ Propellant Flow Rate	82 Lb/sec
○ Fuel Form	Binary (Zr,U)C/NbC
○ Particle Diameter	500 Micron (20 mils)
○ Bed Power Density (ave)	33 MW/l
○ Core Power Density	3.6 MW/l
○ Fuel Volume	47.7 liters
○ Number of Elements	36
○ Safety Shutdown	Central Poison Rod
○ Vessel Diameter	1.03 meters
○ Reactor Fueled Length	92.5 cm
○ Reactor Mass (no recup/ shielding)	5,340 Lb (2,420Kg)

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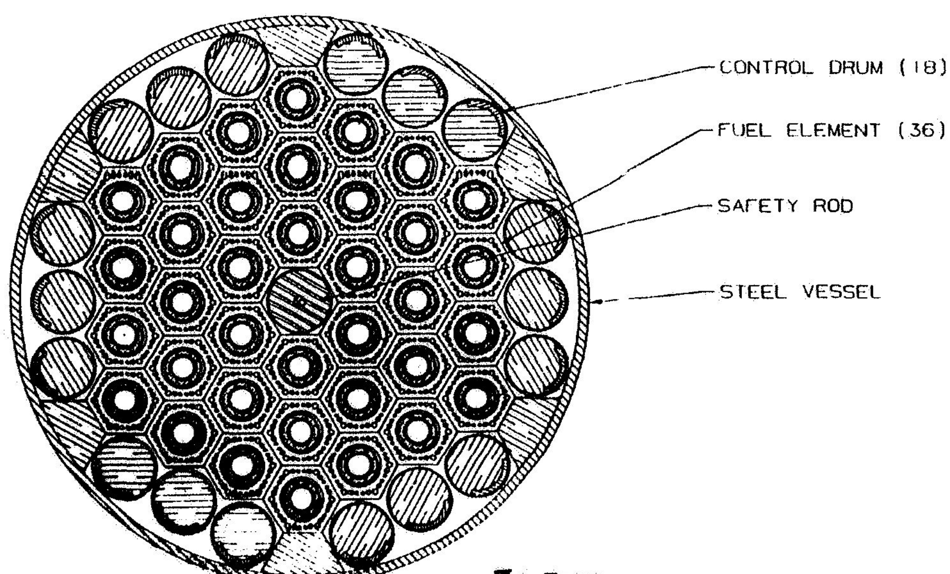
REACTOR FEATURES EFFICIENT INTEGRATION OF COMPONENTS

THE FUEL ELEMENTS ARE LOCATED ON A HEXAGONAL ARRAY. THIS ALLOWS THE 36 FUEL ELEMENTS TO BE EFFICIENTLY INTEGRATED INTO THE SMALLEST POSSIBLE VOLUME WHILE MEETING CRITICALITY LIMITS,THERMAL HYDRAULIC AND STRUCTURAL REQUIREMENTS. A SATISFACTORY PITCH WAS FOUND TO BE 11 CM. THIS WAS PRIMARILY SIZED FOR HYDRAULIC CONSIDERATIONS (IE LOW PRESSURE DROP THROUGH THE MODERATOR AND INLET PLENUM). FURTHER STUDIES AND OPTIMIZATION WILL LIKELY RESULT IN A CHANGE IN THE PITCH.

THE CONTROL DRUMS ACT AS THE REFLECTOR AND THE CONTROL SYSTEM FOR THE REACTOR. THEY ARE APPROXIMATELY 11 CM IN DIAMETER WITH AN OUTER SEGMENT OF 12 MM THICKNESS AND 120 DEGREE ARC CONTAINING B_4C. THEY ARE PLACED AS CLOSE TO THE CORE AS POSSIBLE. ORIGINALLY THERE WERE 24 CONTROL DRUMS BUT THE "CORNER" SIX WERE PROVIDING LITTLE CONTROL BENEFIT AND WERE THEREFORE REMOVED. THE SAFETY ROD IS LOCATED IN THE CENTER OF THE CORE WHERE ITS WORTH IS MAXIMIZED.

keactor Features Efficient Integration of Components



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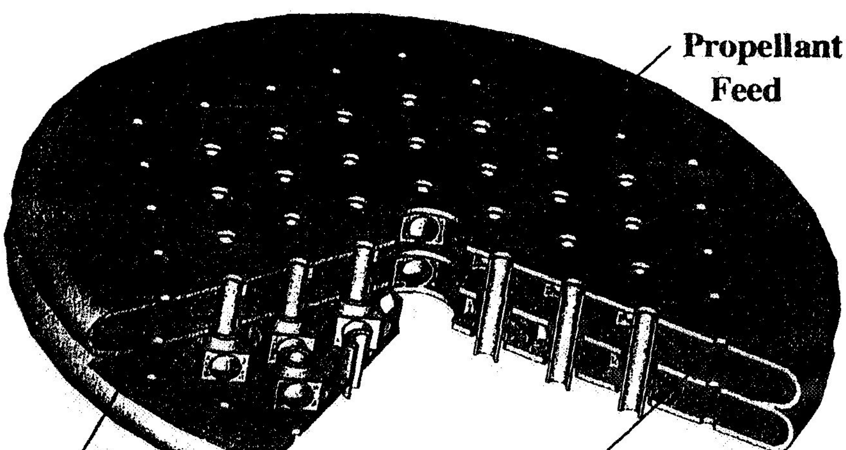
MANIFOLDS PROVIDE CORE SUPPORT

THE VARIOUS FLOW LOOPS OF THE ENGINE SYSTEM HAVE BEEN GREATLY SIMPLIFIED THROUGH THE USE OF AN INNOVATIVE MANIFOLD/CORE SUPPORT STRUCTURE. THIS COMPONENT IS UNIQUE IN THAT IT NOT ONLY PROVIDES A VERY RIGID STRUCTURE TO WHICH THE FUEL ELEMENTS ARE ATTACHED BUT IT ALSO CONTAINS TWO PLENUMS FOR THE MODERATOR COOLING LOOP AND A FEED-THROUGH FOR THE FUEL ELEMENT PROPELLANT LOOP.

THE FABRICATION OF THE CORE SUPPORT STRUCTURE IS SIMILAR TO THAT OF A HONEYCOMB COMPOSITE. THE INTERNAL WEBS CARRY THE SHEAR LOADS WHILE THE TOP AND BOTTOM SKINS OF STEEL CARRY THE MEMBRANE LOADS. AS THE AIRCRAFT INDUSTRY IS AWARE, THIS CONFIGURATION IS EXTREMELY EFFICIENT IN ITS SPECIFIC LOAD CARRYING CAPABILITY AND STIFFNESS, THEREFORE THE THICKNESSES OF THE STEEL SKINS AND WEB ARE MINIMIZED.

MANIFOLDS PROVIDE CORE SUPPORT



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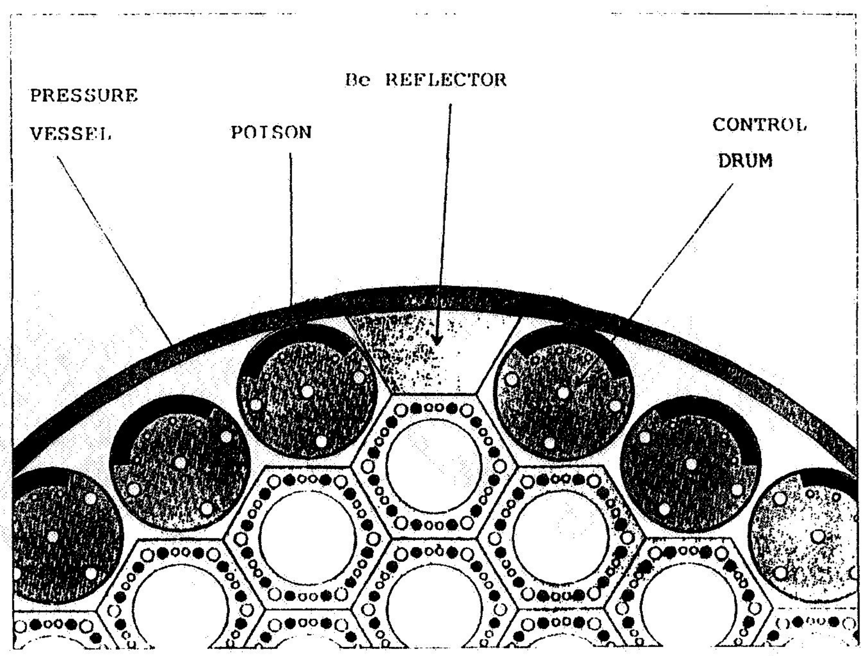
CONTROL DRUMS POSITIONED FOR MAXIMUM WORTH

THE CONTROL DRUMS ARE LOCATED CLOSE TO THE CORE, WITH AS LARGE A DRUM SIZE AS POSSIBLE WITHOUT INTERFERENCE, (APPROXIMATELY 11 CM OUTSIDE DIAMETER) TO ENHANCE NEUTRON REFLECTION AND CONTROL WORTH, WHILE MINIMIZING THE SURROUNDING PRESSURE VESSEL SIZE. THE DRUM HEIGHT IS SLIGHTLY LONGER THAN THE ACTIVE FUEL BED HEIGHT OF ABOUT 92 CM. THE CONTROL DRUMS FOR THE CONCEPTUAL DESIGN ARE MADE OF BERYLLIUM WITH A B_4C POISON SEGMENT 12 MM THICK OVER A 120 DEGREE ARC SEGMENT. THE CONTROL DRUM WORTH IS 0.10 DELTA-K/K WITH THE SAFETY ROD WITHDRAWN, AND 0.14 WITH THE SAFETY ROD INSERTED. TOTAL CONTROL WORTH IS 0.26 DELTA-K/K, FOR A SHUTDOWN REACTIVITY OF -0.20 DELTA-K/K (K-EFFECTIVE = 0.83). NOMINAL HYDROGEN GAS DENSITIES, OR WORTH IS 0.07 DELTA-K/K, WERE INCLUDED. THUS, WITHOUT HYDROGEN GAS, THE REACTOR WILL BE 0.01 DELTA-K/K SHUTDOWN EVEN IN THE MOST REACTIVE CONTROL POSITIONS. A STUDY OF INDIVIDUAL DRUM WORTHS SHOWED THAT A FIXED BERYLLIUM REFLECTOR SECTION IN THE CORNER POSITION ENHANCES REFLECTION AND CONTROL WORTH, INCREASING TOTAL DRUM WORTH BY ABOUT 0.01 DELTA-K/K, WHILE ALSO ALLOWING SMALLER PRESSURE VESSEL SIZE. A TRADE STUDY SHOWED POISON THICKNESS AS THIN AS 1 MM IS QUITE EFFECTIVE AND THAT THE MAXIMUM WORTH WAS REACHED AT APPROXIMATELY 10 TO 15 MM.

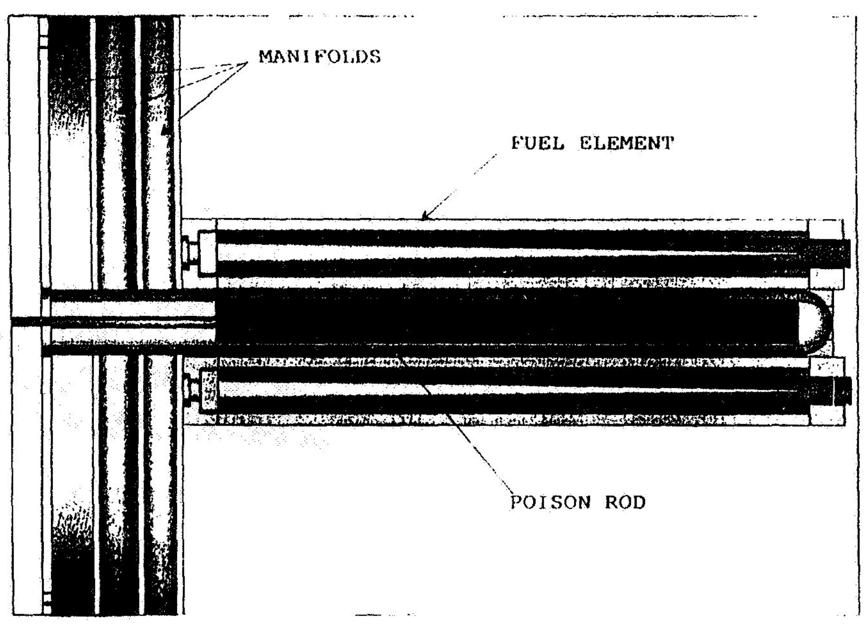
CONTROL DRUMS POSITIONED FOR MAXIMUM WORTH



SAFETY ROD LOCATED FOR MAXIMUM WORTH

THE CENTRAL SAFETY ROD IS LOCATED IN A POSITION OF HIGH NEUTRON FLUX, RESULTING IN A CONTROL WORTH WHICH EXCEEDS THE COMBINED WORTH OF THE 18 CONTROL DRUMS. IT CONTAINS B_4C IN A CYLINDRICAL SHAPE WITH OUTSIDE DIAMETER OF ALMOST 11 CM. A BeO REFLECTOR SEGMENT IS MOUNTED ON THE AFT END TO MINIMIZE NEUTRON STREAMING THROUGH THE SAFETY ROD OPENING AND TO REDUCE HEATING DURING OPERATION. THE SAFETY ROD WORTH IS 0.12 DELTA-K/K WITH THE CONTROL DRUMS' POISON OUTWARD, AND 0.16 DELTA-K/K WITH THE CONTROL DRUMS' POISON INWARD. THE TOTAL CONTROL WORTH IS 0.26 DELTA-K/K, FOR A SHUTDOWN REACTIVITY OF -0.20 DELTA-K/K (K-EFFECTIVE = 0.83). NOMINAL HYDROGEN GAS DENSITIES, WORTH 0.07 DELTA-K/K, WERE INCLUDED. THUS, WITHOUT HYDROGEN GAS, THE REACTOR WILL BE 0.01 DELTA-K/K SHUTDOWN EVEN IN THE MOST REACTIVE CONTROL POSITIONS.

SAFETY ROD LOCATED FOR MAXIMUM WORTH



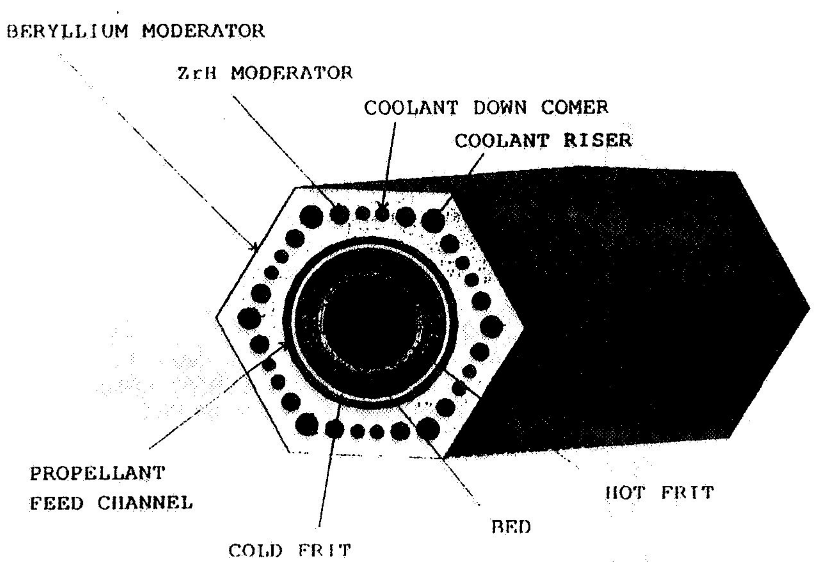
FUEL ELEMENT INTEGRATED EFFICIENTLY

THE PBR CORE IS HETEROGENEOUS WITH A COOLED MODERATOR JACKET SURROUNDING THE COLD FRIT, FUEL AND HOT FRIT. THE MODERATOR CONSISTS OF MACHINED BLOCKS OF BERYLLIUM JOINED TOGETHER TO FORM A RIGID JACKET NEARLY 1 METER IN LENGTH. THE BLOCKS ARE PRE-DRILLED WITH CAREFULLY SIZED HOLES. TWELVE OF THE 36 HOLES ARE FILLED WITH ZIRCONIUM HYDRIDE AND THE REMAINDER ARE USED FOR MODERATOR COOLING. THE RATIOS OF ZIRCONIUM HYDRIDE, BERYLLIUM AND HYDROGEN YIELD PROPER NEUTRONICS, STRUCTURAL AND THERMAL HYDRAULIC PERFORMANCE. THE BASELINE DESIGN UTILIZES A RATIO OF 82%Be, 8%ZrH AND 10%H_2. THE PRESSURE DROP WITHIN THE COOLANT LOOP OF THE MODERATOR IS APPROXIMATELY 300 PSI, THE THICK BERYLLIUM WALLS ARE MORE THAN ADEQUATE TO PROVIDE STRUCTURAL SUPPORT AND THE ZIRCONIUM HYDRIDE, EVEN IN SUCH SMALL QUANTITIES PROVIDES ENHANCED MODERATION FOR THE CORE. FURTHER OPTIMIZATION CAN SIGNIFICANTLY ENHANCE THE PERFORMANCE OF THE MODERATOR AND REACTOR.

THE COLD FRIT DISTRIBUTES THE FLOW TO THE FUEL BED IN PROPORTION TO THE LOCAL HEAT GENERATION. COOLANT FLOW IS DIRECTED RADIALLY INWARD AND IS HEATED BY THE FUEL BED. THE HOT GAS PASSES THROUGH HOLES IN THE HOT FRIT WHERE IT COLLECTS IN THE HOT CHANNEL AND IS EXHAUSTED TO THE NOZZLE. THE COLD FRIT IS MADE OF STEEL AND WILL LIKELY BE OF PLATELET DESIGN. THE HOT FRIT IS MADE OF NIOBIUM CARBIDE COATED GRAPHITE. GRAPHITE TECHNOLOGY HAS EVOLVED OVER THE YEARS AND IT IS NOW POSSIBLE TO OBTAIN GRAPHITE WITH A THERMAL EXPANSION COEFFICIENT THAT MATCHES THAT OF NIOBIUM CARBIDE EXACTLY

THE HOT CHANNEL (THE AREA INSIDE THE HOT FRIT) IS SIZED SUCH THAT THE MAXIMUM VELOCITY OF THE HOT HYDROGEN IS NOT GREATER THAN MACH 0.25 TO AVOID COMPRESSIBILITY EFFECTS

FUEL ELEMENT INTEGRATED EFFICIENTLY



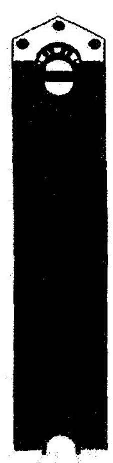
FUEL ELEMENT FLOW IS SELF CONTAINED

THE FUEL ELEMENT WAS ANALYZED USING FOTVE, A 1-D B&W PROPRIETARY COMPUTER CODE WHICH PREDICTS PRESSURE LOSSES IN THE DIFFERENT COMPONENTS OF AN ELEMENT, AS WELL AS TEMPERATURES OF THE PROPELLANT AND FUEL. EACH FUEL ELEMENT WAS INITIALLY ALLOTTED A PRESSURE DROP OF 500 PSID. RESULTS FROM FOTVE INDICATE MAXIMUM PRESSURE DROPS OF ABOUT 300 PSID. THE PROPELLANT FEED CHANNEL WAS DESIGNED TO MINIMIZE PRESSURE LOSS OVER THE LENGTH OF THE CHANNEL WITHOUT IMPACTING THE WEIGHT AND SIZE OF THE FUEL ELEMENT ASSEMBLY. RESULTS OF A STUDY TO COMPARE PRESSURE LOSSES TO THE DIFFERENT GAP SIZES BETWEEN THE MODERATOR AND COLD FRIT INDICATED THAT THE OPTIMUM GAP WAS AT 0.3 CM.
THE COLD FRIT IS DESIGNED TO BE THE PRIMARY FLOW CONTROLLER. IN FOTVE RUNS, THE MINIMUM COLD FRIT PRESSURE DROP TO THE BED PRESSURE DROP RATIO IS MAINTAINED AT 8 TO 1. THIS RATIO ENSURES THAT THE COLD FRIT HAS CONSIDERABLY MORE CONTROL OVER THE FLOW DISTRIBUTION TO THE FUEL BED THAN THE PROPELLANT FEED CHANNEL OR THE BED ITSELF.

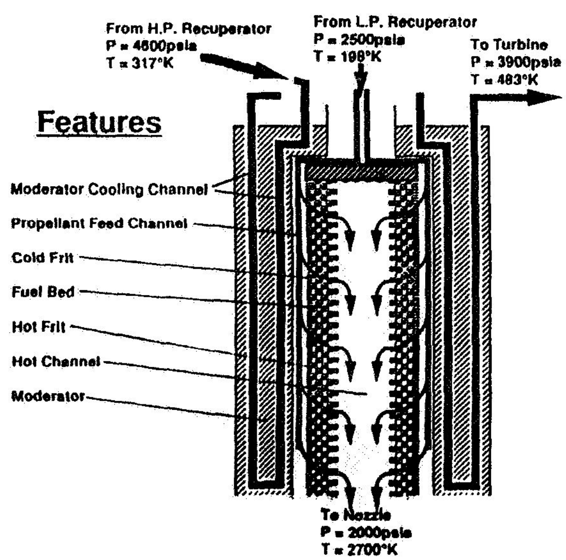
THE MODERATOR DESIGN WAS ANALYZED USING A CHANNEL FLOW CODE CALLED PIPTH, ANOTHER B&W PROPRIETARY CODE. THE CODE CALCULATES PRESSURES, TEMPERATURES AND FILM COEFFICIENTS ALONG THE LENGTH OF A HEATED CHANNEL. THE MODERATORS WERE INITIALLY ALLOTTED A PRESSURE DROP OF 500 PSID. RESULTS INDICATE A MAXIMUM DROP OF ABOUT 300 PSID.

THIS CORE DESIGN HAS NOT BEEN OPTIMIZED. HOWEVER, A FUEL ELEMENT AND MODERATOR FROM ONE OF THE SIX ASSEMBLIES WHICH SURROUND THE SAFETY ROD WERE ANALYZED FOR THIS CORE CONFIGURATION. THESE ANALYSES DEMONSTRATE A WORKABLE DESIGN BUT DETAILED ANALYSES MUST BE PERFORMED ON A CORE SYSTEM LEVEL TO PROVIDE INSIGHT ON FLOW SPLIT CHARACTERISTICS AND ITS IMPACT ON PRESSURES AND TEMPERATURES FOR FULL POWER, THROTTLING AND DECAY HEAT CONDITIONS.

FUEL ELEMENT FLOW IS SELF CONTAINED



Fuel Element Cross-Section (with Afterburner)



FUEL PARTICLE PROVIDES HIGH POWER DENSITY & TEMP

FUEL PARTICLE DESIGN IS BASED ON MISSION REQUIREMENTS AND, WITHIN LIMITS, DOES NOT SIGNIFICANTLY AFFECT THE REACTOR DESIGN IN GENERAL. PARTICLE DESIGN IS SELECTED ON THE BASIS OF ENGINE BURN DURATION, NUMBER OF CYCLES, EXHAUST GAS TEMPERATURE AND SYSTEM RELIABILITY REQUIREMENTS. CURRENTLY, THERE ARE FOUR COATED PARTICLE DESIGNS UNDER CONSIDERATION FOR USE IN PBR'S, EACH CAPABLE OF MEETING DIFFERENT SETS OF MISSION REQUIREMENTS.

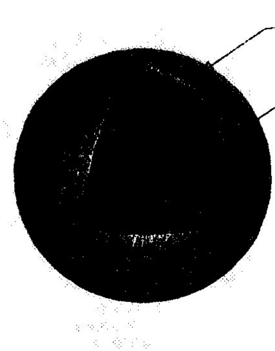
1. THE EARLIEST PARTICLE B&W DEVELOPED WAS THE SO-CALLED BASELINE PARTICLE. IT WAS BASED ON THE TRISO PARTICLE DEVELOPED FOR GAS-COOLED POWER REACTORS. IT CONSISTS OF A URANIUM CARBIDE KERNEL SURROUNDED BY TWO LAYERS OF CARBON (BUFFER, FOR CTE MISMATCH AND SEALANT) AND AN OUTER SHELL OF ZrC OR NbC. THIS PARTICLE IS CAPABLE OF OPERATING FOR 10'S TO 100'S OF SECONDS IN THE RANGE OF 2500-2800K FOR 5-10 THERMAL CYCLES. THESE PARTICLES HAVE BEEN PRODUCED BY B&W IN SIGNIFICANT QUANTITIES AND THE PROCESS IS WELL UNDERSTOOD. TESTING HAS INCLUDED BOTH IN-PILE AND OUT-PILE TESTS. THIS PARTICLE WOULD NOT SUSTAIN 2700K FOR SEI APPLICATIONS.

2. WE ARE PRESENTLY DEVELOPING MIXED CARBIDE PARTICLES WHICH ARE DESIGNED TO REACH MAX. FUEL TEMPERATURES OF 3200-3400K. THEY ARE DESIGNED TO OPERATE FOR 100'S TO 1000'S OF SECONDS. MIXED CARBIDES SUCH AS (U,Zr)C AND (U,Nb)C WITH NbC COATINGS ARE BEING DEVELOPED. IT IS EXPECTED THAT THIS PARTICLE WILL WITHSTAND MORE THAN 10 THERMAL CYCLES AND SUSTAIN 2700 K EXIT GAS TEMPERATURE OR GREATER. THESE ARE CONSIDERED THE APD/B&W BASELINE SEI FUEL. B&W EXPECTS TO BE IN FULL PRODUCTION OF THESE PARTICLES WITHIN 1 YEAR.

3. UNDER REVIEW IS A PARTICLE DESIGN CONSISTING OF A POROUS UO_2 KERNEL COATED WITH TUNGSTEN. B&W EXPECTS IT TO BE EASILY FABRICABLE, HAVE GREATER LONG TERM (1000'S OF SECONDS) RELIABILITY THAN CARBIDE FUELS, BE VERY RESISTANT TO THERMAL CYCLES IF KEPT ABOVE ABOUT 700 K, AND HAVE VERY GOOD FISSION PRODUCT RETENTION BUT BE LIMITED TO A MAXIMUM TEMPERATURE OF ABOUT 3000-3100K.

4. IN THE EARLY STAGES OF LAB DEVELOPMENT ARE ADDITIONAL ADVANCED PARTICLE CONCEPTS WHICH ARE THEORETICALLY CAPABLE OF OPERATING FOR 1000'S SECONDS AT 3400K AND MANY THERMAL CYCLES. IT IS EXPECTED THAT THESE COULD BE BROUGHT TO PRODUCTION READINESS IN SEVERAL YEARS.

FUEL PARTICLE PROVIDES HIGH POWER DENSITY & TEMP



- **48 Liters of Fuel**
- **33 MW/l (ave)**
- **500 Micron Dia.**
- **Loading:**
  - 127 kg Uranium
  - 21 kg Carbon
  - 57 kg Zirconium
  - 60 kg Niobium
  - **265 kg Total Bed**
- **73% Enrichment**
- **3,300+ K Melt Approx.**

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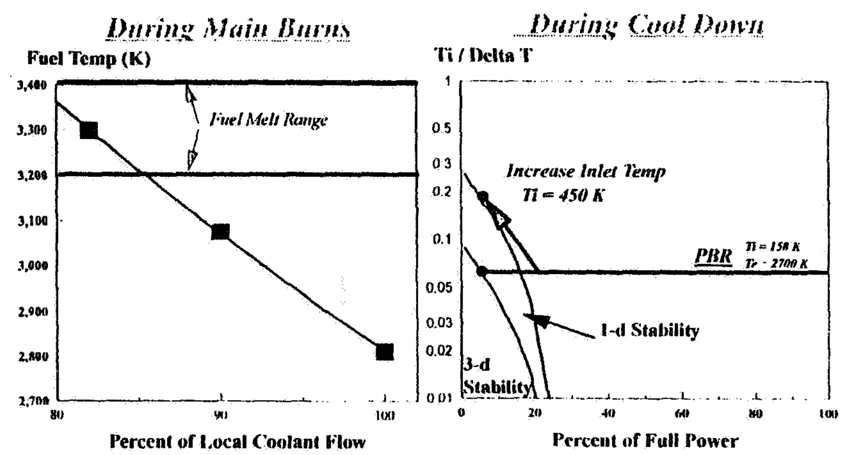
THE PBR IS THERMALLY STABLE. THE FUEL TEMPERATURE OPERATES AT ABOUT 2800K WHEN THE MIXED MEAN GAS TEMPERATURE IS 2700K. THE AMOUNT OF LOCAL FLOW CAN BE REDUCED BY 15-22 % BEFORE THE FUEL KERNEL REACHES ITS MELT TEMPERATURE. BASED ON THERMAL HYDRAULIC STABILITY STUDIES A LOCAL FLOW DISRUPTION DOES NOT CAUSE A PROPAGATION BUT RATHER A STABLE TRANSITION TO A NEW TEMPERATURE. THIS IS DUE TO THE HIGH REYNOLDS NUMBER AND TURBULENT REGIME OVER WHICH THIS PBR OPERATES. FOR HIGH POWER REACTOR OPERATION THE PBR IS QUITE STABLE (AS IS EXPECTED).

THERMAL HYDRAULIC INSTABILITY CAN OCCUR FOR LOW FLOW REGIMES (IE. LOW POWER). THE BED HYDRAULIC RESISTANCE, WHICH IS FORMED BY VISCOUS AND INERTIAL FORCES (TYPIFIED BY THE ERGUN CORRELATION) IS DOMINATED BY INERTIAL FORCES FOR HIGH POWER, HIGH FLOW OPERATION. HOWEVER, FOR LOW FLOW OPERATION THE VISCOUS TERM CAN DOMINATE. FOR SUCH CASES, A PERTURBATION IN FLOW CAN CAUSE INCREASED LOCAL GAS TEMPERATURE AND THUS HIGHER PRESSURE WHICH CAUSES HIGHER GAS TEMPERATURES...AND SO ON. B&W UNDERSTANDS THE MECHANISMS INVOLVED AND THE REGIMES OF OPERATION WHICH MUST BE AVOIDED. IT IS SHOWN THAT FOR A THREE DIMENSIONAL ANALYSES IT IS POSSIBLE TO RETAIN NEARLY ALL THE PBR PERFORMANCE (IMPULSE) WHILE THROTTLING TO 5-10% OF FULL FLOW. ANOTHER SOLUTION TO THE LOW FLOW INSTABILITY CAN BE FOUND BY REDUCING THE CHANGE IN GAS TEMPERATURE AS IT FLOWS THROUGH THE FUEL BED. BY INCREASING THE INLET GAS TEMPERATURE TO 450K, THE FLOW STABILITY CAN BE MAINTAINED EVEN FOR CONSERVATIVE ANALYSES.

The PBR NTRE is Thermally Stable



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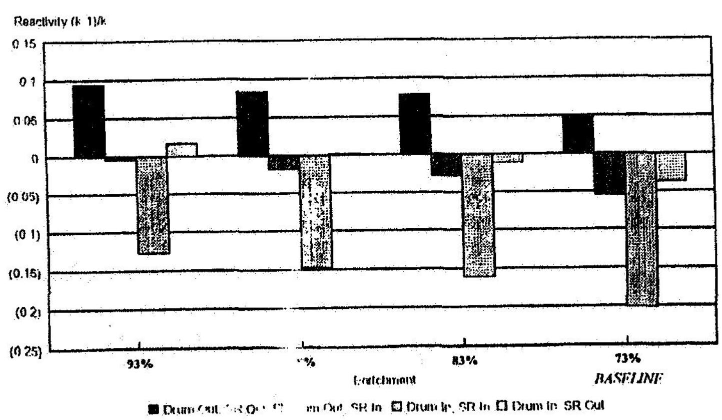
FUEL DESIGN BALANCES REACTIVITY AND CONTROL

THE REACTOR AND FUEL DESIGN IS FLEXIBILE. THE 48 LITER FUEL VOLUME WAS BASED ON 33 MW/LITER POWER DENSITY AND 1566 MW TOTAL POWER. THE 36 FUEL ELEMENT CORE WITH 11 CM PITCH AND MODERATOR COMPOSED OF 82% BERYLLIUM, 8% ZRH, AND 10% HYDROGEN COOLANT PASSAGES WERE BASED PRIMARILY ON MECHANICAL AND THERMO-HYDRAULIC CONSIDERATIONS, AND PHYSICS TRADE-OFFS.

FOR HIGH REACTIVITY, THE FUEL PARTICLE MAXIMIZES URANIUM LOADING. THE BASELINE PARTICLE IS 70% UC/ZRC KERNAL SURROUNDED BY 30% NBC COATING, BY VOLUME, AND THE KERNAL CONTAINS 50% UC, FOR AVERAGE BED URANIUM DENSITY OF 2.7 G/CC. IT IS EXPECTED THE URANUIM LOADING WILL BE REDUCED BY 50% OR MORE THROUGH OPTIMIZATION OF THE CORE. WITH FULLY ENRICHED FUEL, AND NO HIC HOT CHANNEL INSERTS, THE MAXIMUM REACTIVITY IS 0 19 DELTA-K/K. HIGH EXCESS REACTIVITY PROVIDES FLEXIBILITY FOR OPTIMIZING THE DESIGN, AND ALLOWS MARGIN FOR MODELING UNCERTAINTIES AND OVERCOMING REACTIVITY LOSSES DUE TO XENON TRANSIENTS AND FUEL DEPLETION DURING OPERATION.

DESIGN VARIABLES TO BALANCE HIGH REACTIVITY AND CONTROL ARE FUEL PARTICLE URANIUM CONTENT, URANIUM ENRICHMENT, INSTALLED NEUTRON POISONS, MODERATOR MATERIALS, FUEL ELEMENT PITCH, AND CORE REFLECTOR AND CONTROLS DESIGN. CONICAL HIC INSERTS WERE PLACED IN THE HOT CHANNELS TO PROVIDE FIXED NEUTRON POISON HOLD-DOWN OF 0.07 DELTA-K/K AND 1% OF TOTAL POWER IN EXTRA HYDROGEN GAS HEATING. HAFNIUM ALSO PROVIDES RESONANCE NEUTRON ABSORPTION WHICH WILL IMPROVE PROMPT TEMPERATURE FEEDBACK; HOWEVER HYDROGEN FLOW DISTRIBUTION MUST COMPENSATE FOR THE SHIFT IN AXIAL POWER SHAPE. THE U^{235} ENRICHMENT WAS ALSO REDUCED, INCREASING THE NEGATIVE PROMPT TEMPERATURE FEEDBACK OF THE FUEL DUE TO THE DOPPLER COEFFICIENT OF THE LARGER FRACTION OF U^{238}. INCREASING PROMPT NUCLEAR FEEDBACK ENHANCES THE STABILITY, AND THUS CONTROL AND SAFETY OF THE REACTOR. THE FINAL MAXIMUM REACTIVITY OF THE CONCEPTUAL DESIGN IS 0.06 DELTA-K/K. THE CONTROLS HAVE LARGE REACTIVITY WORTHS; THE SAFETY ROD WORTH OF 0.12 DELTA-K/K IS SOMEWHAT HIGHER THAN THE CONTROL DRUMS WORTH OF 0.10 DELTA-K/K.

FUEL DESIGN BALANCES REACTIVITY AND CONTROL



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THE PBR IS DESIGNED FOR SAFETY

THE PBR WILL BE MAINTAINED SUBCRITICAL FOR ALL LAUNCH ACCIDENT SCENARIOS. PRELIMINARY ESTIMATES FOUND THAT, FOR THE CONCEPTUAL DESIGN, ADDITIONAL MITIGATING MEASURES ARE NEEDED IN ORDER TO MAINTAIN THE REACTOR SUBCRITICAL IF THE PRIMARY VOID REGIONS, THE COLD AND HOT CHANNELS, ARE FILLED WITH WATER AND THE REACTOR IS SURROUNDED BY WATER TO SIMULATE A WATER IMMERSION ACCIDENT. WHEN THE HOT CHANNELS ARE FILLED WITH TEMPORARY LAUNCH INSERTS MADE OF HfC, FOR EXAMPLE, THE CALCULATED REACTIVITY AFTER IMMERSION IS -0.22 DELTA-K/K (K-EFFECTIVE = 0.82); THE REACTIVITY PRIOR TO IMMERSION IS ABOUT -0.32. FOR THE UNCHANGED BASELINE DESIGN, THE REACTIVITY AFTER IMMERSION WAS 0.07 DELTA-K/K, WHICH IS CLEARLY UNACCEPTABLE. FOR THESE ESTIMATES, THE CHANGES IN THE VOID REGIONS OF THE HOT FRIT, COLD FRIT, FUEL BED, AND MODERATOR COOLING PASSAGES WERE NEGLECTED. IT WILL BE POSSIBLE, THROUGH ADDITIONAL STUDY, TO OBTAIN ACCEPTABLE RESULTS WITHOUT RESORTING TO TEMPORARY LAUNCH INSERTS BY MAKING OTHER MODIFICATIONS. FOR EXAMPLE, WITH CHANGES IN PITCH AND/OR MODERATOR HYDROGEN CONTENT, BALANCED BY DECREASES IN PARTICLE URANIUM CONTENT AND/OR ENRICHMENT, IT WILL BE POSSIBLE TO OBTAIN A MORE NEUTRONICALLY OPTIMUM INITIAL CORE CONDITION, SUCH THAT THE WATER IMMERSION WILL RESULT IN EITHER OVER-MODERATION AND A DECREASE IN REACTIVITY, OR AT LEAST A SMALLER INCREASE IN REACTIVITY. THESE SAME CHANGES WILL ALSO HELP MITIGATE POTENTIAL REACTIVITY INCREASES DUE TO ANY EXCESSIVE HYDROGEN GAS DENSITY IN THE CORE DURING STARTUP OR TRANSIENTS

BASED ON PITCH TRADE STUDIES PERFORMED USING DIFFERENT FUEL BED THICKNESSES, THE IMPACT OF GEOMETRY CHANGES ASSOCIATED WITH A LAUNCH ACCIDENT (E.G. DEFORMATION OR COMPACTION) IS EXPECTED TO BE LESS SEVERE THAN THE WATER IMMERSION. THE NEUTRONIC OPTIMIZATION DISCUSSED ABOVE WILL SERVE TO MITIGATE THIS EVENT AS WELL.

THE PBR IS DESIGNED FOR SAFETY

- **NUCLEAR CRITICALITY**
  - **-Subcritical in Water Immersion**
    - **0.82 Keff using hot frit plugs**
  - **-Two Independent Shut-down Systems**
  - **-Recuperator Prevents Excessive Hydrogen reactivity insertion in core (ie. no LH2)**

- **THERMAL PROTECTION**
  - **-Five Systems to Cool the Reactor**
    - **Twin Turbopumps (70 % full flow capacity ea.)**
    - **One Electrical Pump (Approx. 5 % capacity)**
    - **One Circulation Pump (Several Megawatt cap.)**
    - **Tank Bleed**
  - **-Cross Feed is an Option**

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THE PBR IS SCALABLE IN POWER DENSITY .

THE PBR CAN OPERATE OVER A WIDE RANGE OF POWER DENSITIES. FOR POWER DENSITIES ABOVE 33 MW/L THERE IS ONLY A SMALL IMPROVEMENT IN REACTOR MASS. THE KNEE IN THE CURVE APPEARS TO BE AROUND 20 MW/L. THE LOWER EXHAUST GAS TEMPERATURE (2500K) DOES NOT HAVE A SIGNIFICANT IMPACT ON REACTOR MASS BUT WILL IMPACT PERFORMANCE IN TERMS OF IMPULSE AND THEREFORE HYDROGEN TANKAGE. THE LOWER CHAMBER PRESSURE REACTOR (1000 PSI) APPEARS ATTRACTIVE FROM A REACTOR MASS STANDPOINT, HOWEVER THE REDUCTION IN REACTOR MASS WILL ALMOST CERTAINLY BE NEGATED BY THE INCREASE IN SHIELD MASS SINCE THE VESSEL IS LARGER. THE KEY FEATURES OF THE 33, 20 AND 10 MW/L REACTORS ARE:

SCALABILITY - Power Density

• Bed Power Density	33	20	10	MW/l
• Reactor Power	1,560	1,560	1,560	MW
• Thrust	75,000	75,000	75,000	lb
• Reactor Mass	2,420	2,779	3,687	Kg
• T/W (reactor mass only)	14	12	9	
• Outlet Gas Temp	2,700	2,700	2,700	K
• Fuel Volume	48	79	157	Liters
• Propellant Flow Rate	37	37	37	Kg/s
• Number of Fuel Elements	36	60	60	
• Vessel Diameter	100	106	119	cm
• Reactor Fueled Length	92	92+	92+	cm

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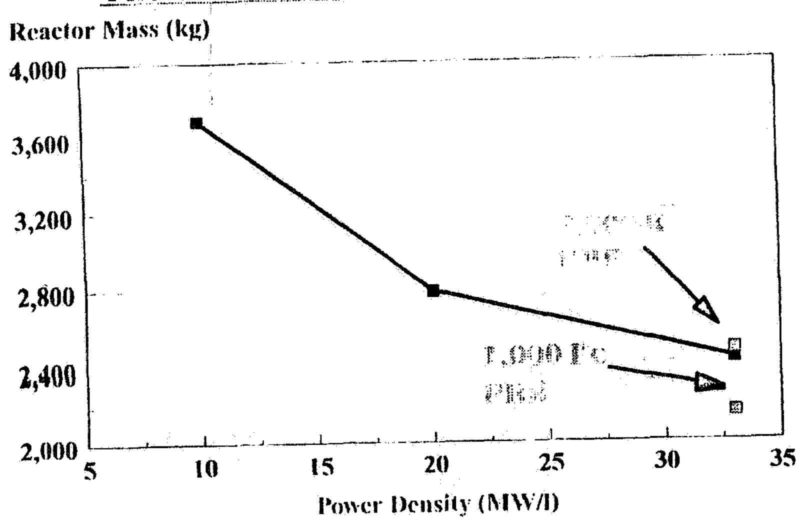
The PBR is Scalable in Power Density



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SCALABILITY - Thrust

Thrust	75,000	40,000	25,000	lb
Reactor Power	1,560	832	520	MW
Reactor Mass	2,420	1,301	1,160	Kg
T/W (reactor mass only)	14	14	10	
Outlet Gas Temp	2,700	2,700	2,700	K
Fuel Volume	48	25	20	Liters
Propellant Flow Rate	37	20	12	Kg/s
Number of Fuel Elements	36	18	18	
Vessel Diameter	100	78	76	cm
Reactor Fueled Length	92	66	66	cm

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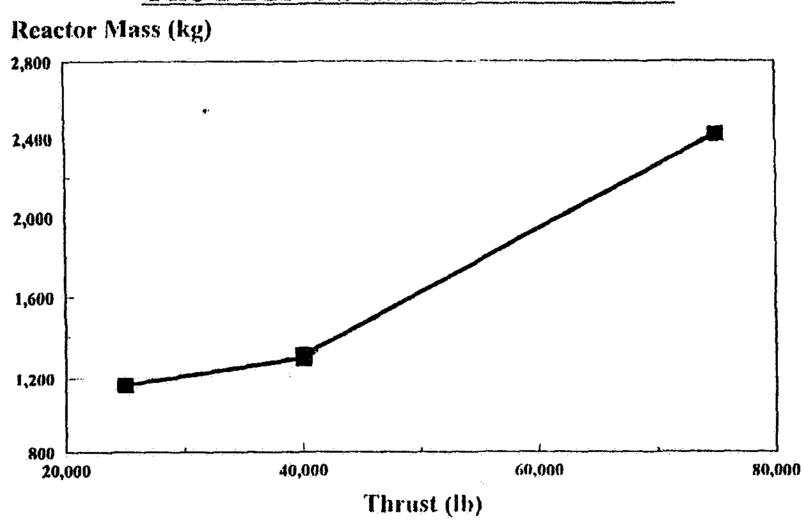
The PBR is Scalable with Thrust



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THE FUEL IS THE NTRE'S KEY TECHNOLOGY

THE PARTICLE BED REACTOR IS UNIQUE BECAUSE OF THE FUEL FORM. ITS SMALL SIZE MEANS THAT IT IS COMPLETELY "PRE-CRACKED". THE DIAMETER OF THE FUEL PARTICLES ARE NEARLY AS SMALL AS THE MANUFACTURING PROCESS WILL ALLOW. OVER 10 MILLION PARTICLES ARE CONTAINED WITHIN EACH OF 36 FUEL ELEMENTS. BECAUSE THE FUEL PARTICLES ARE SO SMALL THE STRESSES WITHIN THE PARTICLES ARE REDUCED AND THEREFORE THE RELIABILITY IS IMPROVED. HIGHLY STRESSED FUELS FORMS WILL FAIL DURING OPERATION.

THE PHYSICAL ARRANGEMENT OF THE FUEL BED IS BENEFICIAL IN TERMS OF FISSION PRODUCT RETENTION. THE OUTSIDE SURFACE OF EACH FUEL BED WILL OPERATE AT MUCH HIGHER POWER DENSITY THAN THE INSIDE SURFACE OF THE BED. THE FISSION EFFICIENCY OF THE INNERMOST PARTICLES IS NOT AS HIGH BECAUSE THE THERMAL NEUTRONS (WHICH WERE THERMALIZED BY THE MODERATOR) DO NOT PENETRATE FAR INTO THE BED. TO INCREASE THE EFFICIENCY OF THE BED AND REDUCE THE "BLACK-NESS", THE BEDS ARE MADE AS THIN (RADIALLY) AS POSSIBLE. IN FACT, THIS SO-CALLED SELF-SHIELDING IS TO A LARGE DEGREE WHAT DETERMINES THE NUMBER OF FUEL ELEMENTS IN THE PBR REACTOR. SINCE THE OUTER PARTICLES ARE PRODUCING THE MOST POWER THEY ALSO PRODUCE THE MOST FISSION PRODUCTS. THE BENEFIT OF THE PBR IS THAT THESE OUTER, HIGH POWER PARTICLES ARE ALSO THE COOLEST SINCE THE COLD GAS COOLS THEM FIRST AS IT MOVES RADIALLY THROUGH THE BED. THE HIGH DIFFUSION RATES TYPICAL AT HIGH TEMPERATURES WILL BE, TO A GREAT EXTENT, HALTED BY THE RELATIVELY COOL PARTICLES. ADDITIONAL FISSION PRODUCT RETENTION CAN EASILY BE DESIGNED INTO THE PARTICLE IN THE FORM OF BUFFER LAYERS OF OTHER COATINGS. THE TRADE IS THE FISSION PRODUCT RETENTION CAPABILITY VS THE URANIUM LOADING OF THE PARTICLE.
FINALLY, THOUSANDS OF INDIVIDUAL PARTICLES CAN FAIL WITH LITTLE OR NO EFFECT ON REACTOR PERFORMANCE WHEREAS MONOLITHIC FUEL FORMS MAY NOT DEGRADE SO GRACEFULLY.

THE FUEL IS THE NTRE's KEY TECHNOLOGY

FUEL FORM AND ARRANGEMENT FAVOR FISSION PRODUCT RETENTION

- **PBR FUEL FORM IS ATTRACTIVE BECAUSE:**
  - Design options permit reduction of fission product retention with additional coatings
  - Individual particle failures have little or no effect on reactor performance
  - Particles have low thermal gradients (small size)

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CONCLUSIONS

THE RESULTS OF THIS WORK INDICATES THAT THE PBR DESIGN MEETS ALL THE NASA REQUIREMENTS. THE RECUPERATED PBR APPEARS TO BE WELL SUITED FOR THE SEI MISSION. THROUGHOUT THE DESIGN PROCESS, TRADES WERE PERFORMED TO FIND APPROPRIATE BLENDS OF SAFETY, RELIABILITY AND STRONG ROBUST COMPONENTS. VERY FEW OPTIMIZATION STUDIES WERE PERFORMED TO EXCEED THE PERFORMANCE REQUIREMENTS BUT IT IS BELIEVED THAT SIGNIFICANT GAINS CAN BE ACCOMPLISHED FROM SUCH OPTIMIZATION. THE PBR TECHNOLOGY APPEARS TO BE CAPABLE OF VERY HIGH PERFORMANCE.

CONCLUSIONS

The PBR Design has been Successfully Adapted for the SEI Mission

- High Performance ***(with mass penalties)***
- Throttling Capability (>>4:1)
- Superior Decay Heat Removal System
- Integrated into Practical Engine Design
- High Reactivity, Control and Safety

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CIS Engine Design

Don Culver

Our CIS engine concept is best summarized by including the rationale behind the selection of each major subsystem concept or operating parameter. 2900 K mixed mean reactor outlet gas temperature is selected to meet the 4.5 hour life requirement with an appropriate margin by the end of fuel assembly development. The engine design/operating selection of 300:1 nozzle area ratio and nozzle inlet pressure of 2,000 psia is the result of a Mars mission payload trade study; it gives the best combination of engine specific impulse and weight. A recuperated turbopump drive cycle was selected for several reasons: (1) nozzle inlet pressures of 1000 psia and above are enabled by recycling topping heat through the turbine, and no reactor manifolding need be added to extract turbine drive heat directly from the core, (2) engine start and shutdown transients are smoothed and assisted by the large, available heat capacity of the high surface area recuperator, (3) the steel heat exchanger adds no weight to the engine, because its weight is determined by its other duties as the forward closure of the reactor vessel. An aft core support structure was selected, because it has been shown by test that CIS fuel assembly life is superior when held in compression. A single DeLaval nozzle is selected that is internally cooled with hydrogen; no hydrogen bleed is necessary, because our formed platelet nozzle operates with low internal wall temperatures at high heat fluxes. A 40 Klbf nozzle of similar design, material and coolant is now in test at NASA. The nozzle is small, because of the engine's high operating pressure, and we use a low weight, carbon-carbon nozzle extension. Its surface may be converted to ZrC to improve its life in hydrogen environment, using near term technology processes similar to those in work at Aerojet, however this is probably unnecessary, because total surface recession in 4.5 hours of operation is expected to be less than 0.025 in. (0.64 mm). A borated ZrH and LiH neutron shield is located within the reactor vessel, between the core and recuperator/gamma shield. It is cooled by propellant flow in steel wafers located between its many transverse layers.

Design Rationale for NTRE With CIS Reactor Is Clear

	Design Parameter	Rationale
1	H_2 Mixed Mean Outlet Temperature (2,900K)	Expected 4.5 Hours Fuel Life (Demoed > 1 Hour at 3000K)
2	Pc (2,000 psia)	Best Mars Mission Performance With Reusable Engines
3	Ae/At (300)	
4	Power Cycle (Recuperated)	• Enable High Pressure With Simple Reactor • Enhances Transient Operation • No Weight Penalty (γ Shield)
5	Core Support (aft))	Optimum With CIS FAs (Test Data)
6	Nozzle (Single DeLaval) (H_2 Cooled Without Bleed Flow) (C-C Extension)	Near Term Technology Use (Formed Cu Platelet)
7	Neutron Shield (Internal)	Lowest Weight and Risk

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Reactor gas outlet temperature for our CIS engine was selected by analyzing fuel assembly in-reactor test data. Fuel assembly tests demonstrated lifetimes of 4000 hours at gas outlet temperatures averaging 2000 K and lifetimes of 4000 seconds at gas outlet temperatures between 3000 K and 3100 K. An Arrhenius law study was applied to the data to predict lifetimes at other outlet temperatures. This work showed that fuel assembly lifetimes of 4.5 hours had been demonstrated at mean outlet gas temperatures of about 2800 K. In-reactor tests did not terminate with destroyed fuel assemblies, however, and Russian scientists have estimated the lifetime demonstrable with a three to five year fuel assembly development program to be about 2.8 hours with 3000 K outlet gas temperature. Arrhenius analysis shows this corresponds to a 4.5 hour life at gas temperatures above 2900 K. Thus, 2900 K nozzle inlet temperature was selected to provide the greatest mission benefit within current NASA life requirements.

CIS Fuel Life Is Expected to Be 4.5 Hours at 2900°K

H_2 Gas Tc, °K	Minimum Achieved Life (Hours)	Maximum Achieved Life (Hours)	Life Expected in 3-5 Years (Hours)
3200	0.400	0.700	1.000
3100	0.655	1.111	1.637
3000	1.111	1.820	2.778
2900	1.954	3.100	4.855
2800	3.579	5.470	8.948
2700	6.856	10.08	17.14
2600	13.80	19.45	34.50
2500	19.40	39.57	73.56
2400	66.67	35.40	166.7
2300	162.4	197.1	406.0
2200	428.7	490.7	1,072
2100	1,242	1,333	3,105
2000	4,000	4,000	10,000
Tc at 4.5 Hours, °K	2,764	2,830	2,914
K	$8{,}574 \times 10^{-8}$	$3{,}789 \times 10^{-7}$	$8{,}574 \times 10^{-8}$
λ	49,132	46,160	49,132

Life = $Ke^{(\lambda/TC)}$ (Arrhenius Law)

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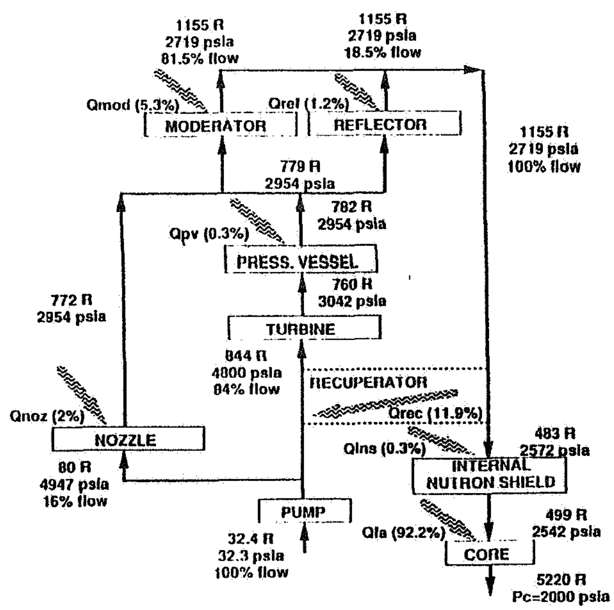
The CIS engine's turbopump is powered by a topping cycle so that no specific impulse is lost through turbine exhaust overboard bleed flows. In this power cycle reactor heat that is deposited in engine components (and must be removed continuously) is removed by the hydrogen propellant, heating pump discharge flow to energy levels high enough to drive the turbines. About 9% of the reactor heat is removed, directly or indirectly, from the engine's nozzle, moderator, reflector, pressure vessel, and radiation shields. However, about 12% of the reactor heat is needed to drive the engine to a nozzle inlet pressure of 2000 psia with adequate (10%) power control margin and reasonable operating conditions for all engine components.

The power cycle requires a liquid hydrogen pump discharge pressure of 4950 psia. We have elected to use two 15,000 horsepower turbopumps in parallel. About 1/6 of the total flow (13 lb/sec) cools the copper nozzle to area ratio 10. The balance of the hydrogen is heated to turbine inlet conditions in the high pressure side of the recuperator (heat exchanger). The turbine inlet temperature is about 850 R (470 K), and the turbine pressure ratio is less than 1.6. Turbine exhaust flow internally cools the walls of the reactor vessel and joins with the nozzle coolant flow at the aft end of the reactor core. here, the 775 R (430 K) flows join and cool the moderator rods and side wall neutron reflector with parallel flows, exiting at about 1150 R (40 K). These gases rejoin and 100% of the propellant flows through the low pressure side of the recuperator. There it loses over 670 R (370 K) temperature to the high pressure flow for turbine drive power. The cool, recuperator outlet gas is distributed to the 102 fuel assembly inlets via the internal neutron shield. Fuel assemblies provide about 93% of the reactor's 1650 MW(t) power to the propellant, so that its maximum exit temperature is 5220 R (2900 K) at a pressure of 2000 psia. This gas flows through the DeLaval rocket nozzle to space.

CIS TOC Recuperated Topping Cycle Flow Schematic



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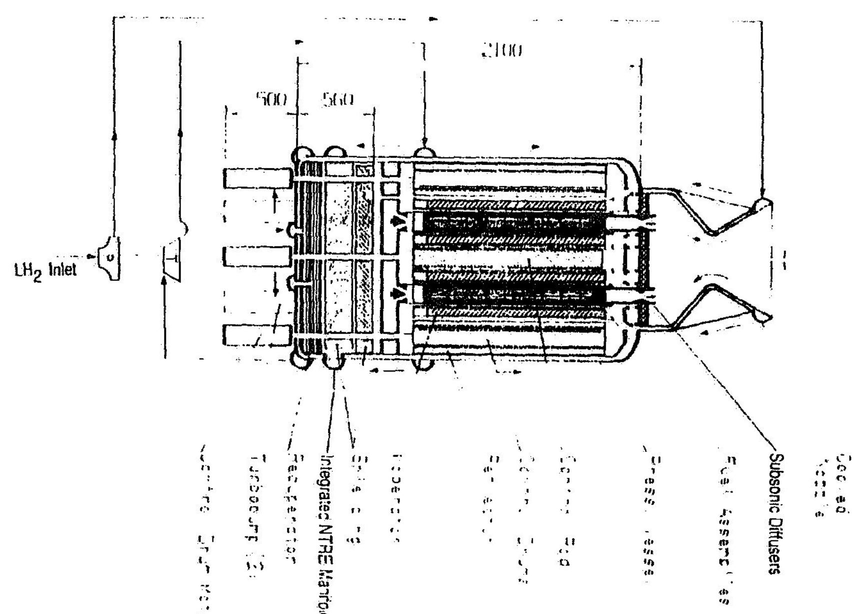
The CIS engine flow scheme begins with two parallel turbopumps. Pump discharge flow splits into two streams; the smaller one cools the copper nozzle through internal passages as in a regeneratively cooled nozzle for bipropellant rocket engines. The nozzle support is cooled by a small portion of this flow. The larger pump discharge flow enters the center of the recuperator/gamma shield located at the forward end of the reactor vessel, distributes to thousands of parallel flow, high pressure passages, and flows radially outward to a peripheral manifold. This heated hydrogen gas flows to and through the turbopump's turbine, the exhaust being routed to an inlet manifold on the aft section of the reactor vessel wall. This flow cools the wall internally as it moves aft to join the nozzle coolant at the aft core support structure. 100% of the propellant flows through this structure, cooling it and the aft peripheries of the 102 fuel assemblies. Propellant is metered forward from the support structure through the reactor's moderator and reflector sections in parallel, collecting in a plenum at the forward end of the core. Here, gas flows radially outward to cooling channels in the forward section of the reactor vessel wall. Propellant flows forward to the periphery of the recuperator, where it enters low pressure, radial inflow heat exchanger passages. Cooled hydrogen leaves the aft face of the recuperator's center and distributes axially through the two layers of the plate-type neutron shields. Propellant flows radially outward through these shields inside of metallic platelets to discharge at the inside wall of the reactor vessel. Hydrogen flows to the fuel assembly inlet plenum and into the 102 inlets, where it is heated to maximum temperature and flows aft into the rocket nozzle inlet and through the nozzle to space.

NTRE Flow Scheme With CIS Reactor



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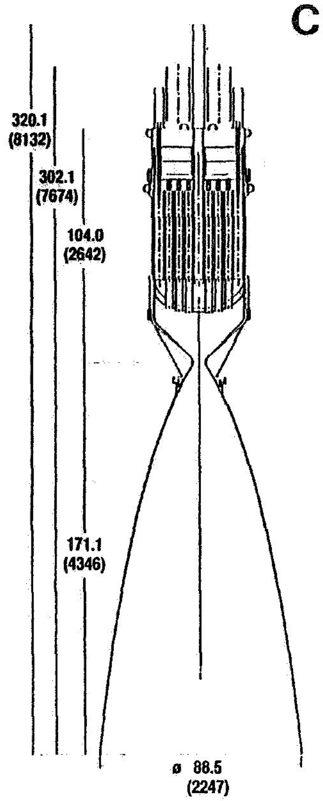
CIS Engine Layout



Thrust, lbf	75000
Chamber Pressure, psia	2000
Nozzle Area Ratio, Ae/At	300
Engine Specific Impulse, sec	959
Mars Mission Specific Impulse, sec	930
Engine Total Weight, lbm	15900
Thrust/Weight	4.7
Engines per Vehicle	2
Payload Returned to Earth, lbm	47067

Engine Weight Breakdown

Component	Weight, lbm
Uncooled Nozzle	232
Cooled Nozzle	965
Pressure Vessel, Reactor Manifolds & CSS	2536
Reactor, Reactor I&C	6613
Turbopump Assemblies (2)	410
Recuperator / Shield	2425
Secondary Shield	642
Plumbing/Valves	1320
Controls and Shielding	758

NTRE w/ Stage Power/Heat Removal

Stage Power & Heat Removal Sys Wt, lbm	2000
Engine with Power Sys Wt, lbm	17900
Mars Mission Specific Impulse, sec	949
Payload Returned to Earth, lbm	52929

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17 69 5c/n/43

CIS Reactor Design

Richard Rochow

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17 69 5c/v44

Reactor Design Summary

- Heterogeneity of the Core
- Average Specific Power Density in FE – 20 MW/1
- Fuel-Elements Twisted Rods on the Base of Solid U-Zr-Nb Carbide Solutions
- Neutron Moderator – Zirconium Hydride Rods
- Controls – 18 Drums in Reflector and 1 Rod in Core
- 7 Safety Rods in Core Against Water Filling Accident
- Reflector – Be
- Internal Shielding – ZrH(B); LiH, Recuperator Steel

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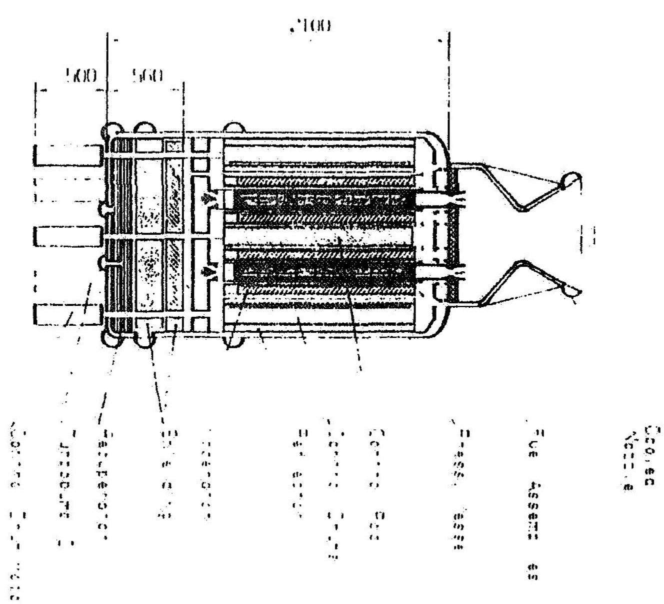
Main Characteristics of NRE Reactor	
Thermal Power, MW	1650
Neutron Spectrum	Thermal
Average Exit Temperature of Propellant, K	2900
Propellant Pressure in Nozzle Chamber, bar	136
Propellant Flow Rate, kg/s	35.4
Reactor Dimensions, mm:	
– Diameter	1050
– Height (including inner shielding)	2100
Mass, kg	5800

NRE Reactor Components Mass (kG)	
Core:	
– FAs	1250
– Moderator	710
Reflector	600
Inner Shielding	
– Zirconium Hydride	430
– Lithium Hydride	120
– Recuperator Material (Steel)	1100
Rotating Drum Drives (18)	360
Safety Rod Drives (7)	80
Supporting Structure	300
Pressure Vessel	850
Total	5800

CIS/NTRE Cross Section



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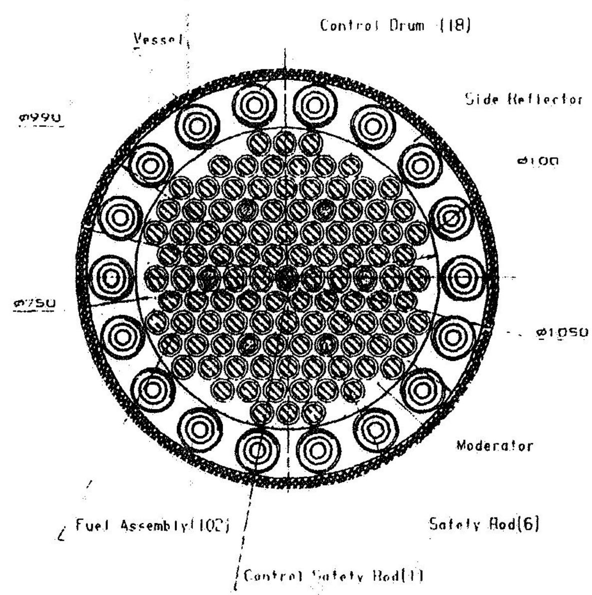
Main Core Parameters

^{235}U Loading, kg	14.1
^{235}U Enrichment, %	90
Average Specific Power Density in FE, MW/liter	20
Non-Uniformity Power Release	
– With the Core Radius	1.2
– With the Core Height	1.2
Thermal Neutron Flux Density, $cm^{-2}.S^{-1}$	1.6×10^{15}
Core Dimensions, mm	
– Diameter/Height	750/1000

CIS/NTRE Reactor Cross Section



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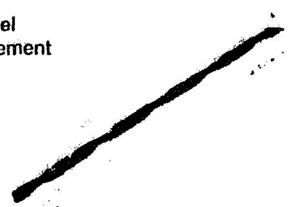
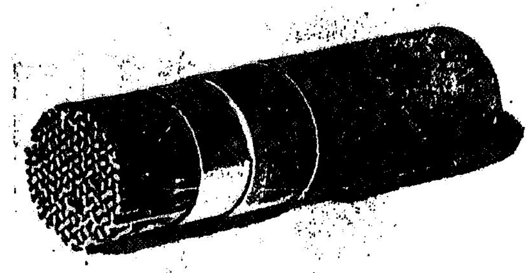
Fuel Assembly Description

Max. FA Thermal Power	**Up to 22 MW**
FE per FA	**356**
Pressure Drop	**40 bar**
Max. Mass Flow Rate	**0.42 kg/s**
FA Dimensions, mm:	
– **Fuel Bundle Length**	**100**
– **Fueled Length**	**1000**
– **Bundle Diameter**	**45**
– **Overall Length**	**1500**
– **Overall Diameter**	**55**
Mass	**12 kg**

Fuel Assembly Performance Exceeds Engine Requirements

Fuel Element





Required Performance Parameters

	T_{H2}, °K	ts	δT/δt	qv MW/L	Pro-pellant	No. (Start)
Aerojet Engine	2900-3000	1000		25	H_2	1...2
In-Pile Tests	3000-3150	1500	~400	25	H_2	3
Max. Parameter Values (in-Reactor)	3100	4000	Up to 1000	35	H_2	12

IVG-1 Performance Parameters

Start-up	T_{H2}, °K	ts	δT/δt	qv MW/L	Pro-pellant	Power, MW
1	3170	~500	100... 150	~22.5	H_2	4.74
2	3110	~500	100... 150	~22.5	H_2	4.81
3	3030	~500	100... 150	~22.5	H_2	4.85

No Fuel Technology Improvements Required

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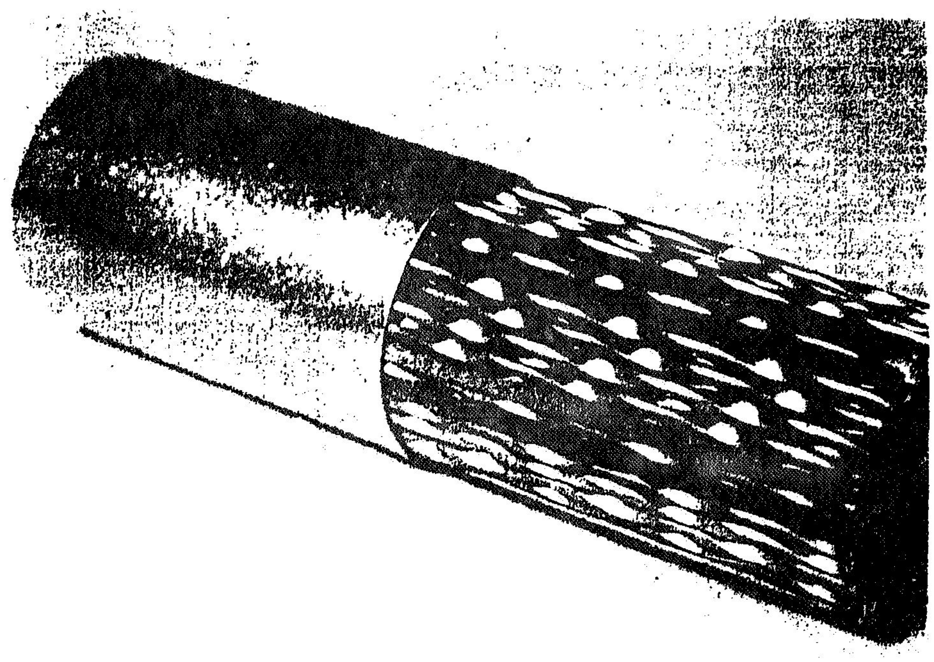
Fuel Element Description

Composition	UC + ZrC + NbC
Max. Fuel Loading of U	Up to 20%
Enrichment	90%
Max. Design Temperature (at Hot End)	3500 K
Operational Temperature (at Hot End)	
– Average	2950 K
– Maximum	3100 K
Average Power Density	
– in FE	20 kW/cm^3
– in FA	12 kW/cm^3
Amount FA Tested at H_2 Above 2,900 K	More than 10,000

Fuel Element Heating Bundle From IVG Reactor



Reactivity and Control Characteristics

1.	Maximum Reactivity Margin, % ΔK	4.2
2.	Reactivity Effects, %	
	– Doppler Effect and Effect of Moderator Temperature	-1.0
	– Hydrogen Filling Effect	+0.6
	– Fuel Burn Up Effect (Compensated by Burning Poisons)	-0.4
	– Water Filling Effect	+7.4
3.	Control System Efficiency, %	
	– 18 Drums	3.0
	– 7 Safety Rods in "dry" Reactor	18.7
	– 7 Safety Rods in "wet" Reactor	8.3
	– Central Rod (in function of regulator)	2.7
4.	Poisoning Material of Controls	B_4C

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AEROJET/ENERGOPOOL/B&W NTRE

NASA LeRC Final Report

Prepared by R.F. Rochow
Babcock & Wilcox, ASE

Oct 23, 1992

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CIS CAPABILITIES

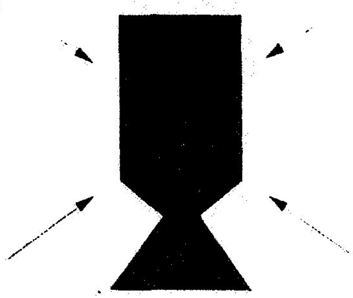
NUCLEAR REACTORS

- **IVG-1 3100K, 240 MW**
- **IRGIT 2650K, 60 MW**
- **IGR 3100K, 35 MW/l**
- **Critical Reactors**
- **Shielding Test Reactors**
- **Materials Test Reactors**

DESIGN

- **250,000 Man-years of NRE Design/Test**
- **30 Years Experience**

NRE for Mars



MANUFACTURING

- **Fuel Line 2 Cores/yr**
- **Insulating Matl's ZrC,NbC**
- **Bulk Fabrication ZrH, LiH**
- **Single Crystal Technology**

TEST FACILITIES

- **Baikal-1 IVG-1, IRGIT**
- **Plasmatron 100 MW**
- **Creep Test Rig 200 kW**
- **Corrosion Tester 250 kW**
- **Failure Mode Rig 100 kW**

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CIS REACTOR DESIGN PHILOSOPHY

- Heterogeneous Core
- Solid Carbide Solution "twisted ribbon" Fuel Elements
- Zirconium Hydride Moderator Rods
- ZrH(B) and LiH Internal Shielding
- Core Support at Hot End
- 12 MW/l Ave. Fuel Bundle Power Density

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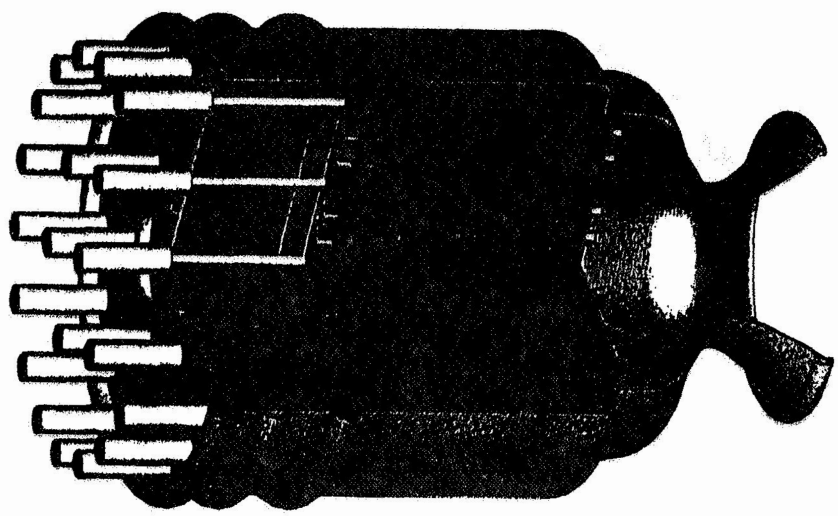
The CIS Reactor Utilizes Demonstrated Hardware



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REACTOR SUMMARY: *KEY SPECS*

• Reactor Power	1.65 GW
• Thrust (200:1 nozzle)	75,000 Lb
• Gas Outlet Temp (mixed mean)	2,900 K (4,730 F)
• Propellant Flow Rate	79 Lb/sec
• Specific Impulse	959 seconds
• Fuel Composition	(U,Nb,Zr)C
• Fuel Form ("Twisted Ribbon")	100x1.6x1.0 mm Approx
• Fuel Bundle Power Density (ave)	12 MW/l
• Core Power Density	3.7 MW/l
• Fuel Volume	162 liters
• Number of Assemblies (Elements)	102
• Safety Shutdown	6 Safety Rods
• Vessel Diameter	1.05 meters
• Reactor Fueled Length	100 cm
• Reactor Mass (no recup/shielding)	4,150 kg (9,150 lb)

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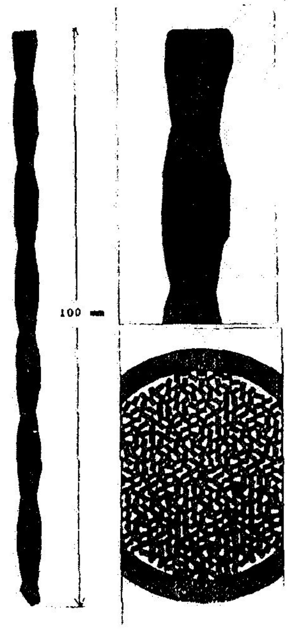
FUEL ELEMENT DESIGN HAS BEEN REFINED



NUMEROUS MATERIALS AND FUEL FORMS TESTED
- *Carbides*
- *Carbonitrides*
- *W, Mo, Re*
- *Graphite Carbides*
- *Particles (1-8 mm Dia)*

MORE THAN 10,000 "TWISTED RIBBONS" TESTED ABOVE 2,900 K

3,000 K FOR THOUSANDS OF SECONDS IN H2- *DEMONSTRATED*

Fuel Element Specifications

Composition	(U,Zr,Nb)C
Max. Uranium Loading	20%
Enrichment	90 %
Max. Design Temp.	3,500 K
Max. Operating Temp.	3,100 K
Power Density:	
- in Fuel Element	20 MW/l
- in Fueled Volume	12 MW/l

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FUEL ASSEMBLY DESIGN HAS BEEN REFINED

FUEL COMPOSITION IS TAILORED
- *axially and radially*
- *for mechanical properties*

INSULATION HAS BEEN DEVELOPED
- *Monolitic NbC and ZrC tubes*
- *Temp capability up to 3,100 K*
- *greater than 50 % porosity*

HUNDREDS OF ASSEMBLIES TESTED

Fuel Assembly Specifications

Max. Thermal Power	22 MW
Pressure Drop Nom.	40 bars
Mass Flow (max.)	0.42 kg/s
Mass	12 kg
Dimensions	
Fueled Length	1.0 m
Fueled Diameter (bundle)	45 mm
Overall Length	1.5 m
Overall Diameter	55 mm

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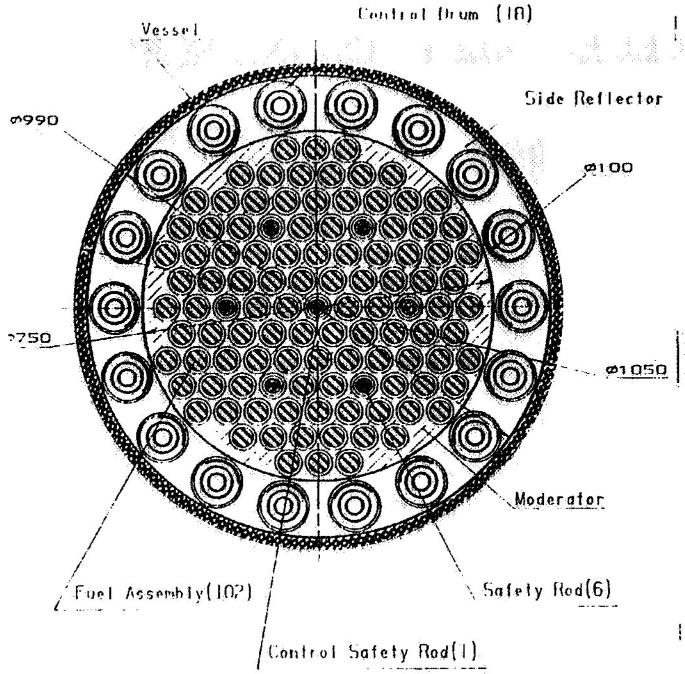
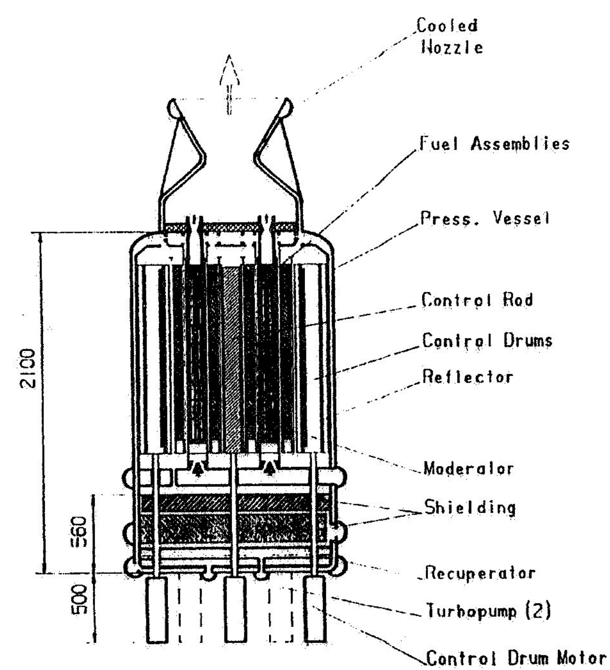
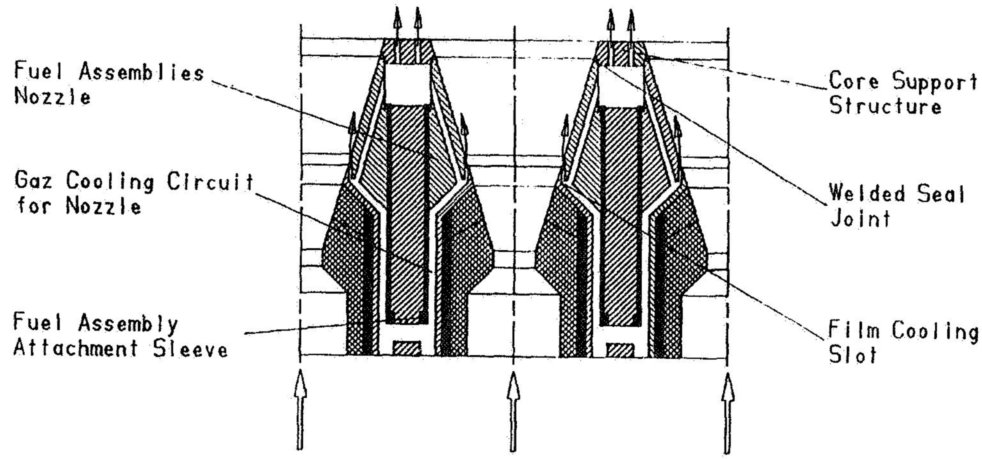


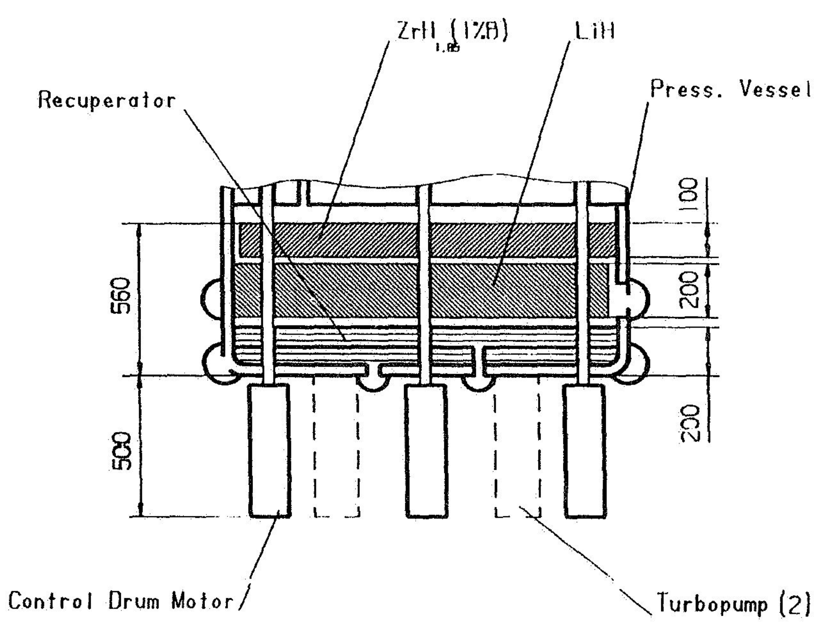
REACTOR RADIAL CROSS-SECTION



REACTOR LONGITUDINAL CROSS-SECTION



CORE SUPPORT STRUCTURE



INTERNAL SHIELD CONFIGURATION

Technology Roadmap

- Demonstrated Reactor Technologies
- Technology Schedules

Mel Bulman

Energopool companies have tested NTRE fuel assemblies in reactors with hydrogen outlet temperatures as high as 3100 K. At the highest temperature the assemblies have been tested successfully for 4000 seconds including 12 starts or thermal transients. Thermal transient rates of about 400 K/sec have been used, but a single start-up rate as high as 1000 K/sec was observed without incident. Fuel assemblies have been shown to resist long duration vibration and high impact loads without critical damage. Fuel power density has been demonstrated to 25 Watt/liter, or about 15 Watt/liter of fuel bundle volume.

Based on results to date, Russian scientists estimate that their fuel assemblies will be able to demonstrate better performance and life within a 3 to 5 year demonstration program. Such improvements can lead to higher performance, lower weight, longer life, rocket engines.

Russia Has Fuel Elements and Fuel Assemblies for NRE Reactor and Plans to Improve Them

Reactors / Parameters	NERVA	IGV-1 and IGR		
	Nerva Reactor	Achieved	Expected In 3-5 Years	Future
Hydrogen Temperature, °K	2550	3000	≥ 3,200	
Specific Impulse, S	850	975	> 1000	
Life-Time, Seconds	3600	4000 s in Hydrogen at T = 3000°K	~10,000 s in Hydrogen at T = 3000°K	~ 30,000 s in Hydrogen +SC at T ≥ 2800°K
Power Density, Mw/Litre	2.5	25	40	80
H_2 Temperature Transient, °K/sec		400		~ 1,000
Number of Starts-Ups	12	12	20	
Vibro-Strength g/Frequency (Hz)/ Testing Time (h)		15/ Up to 3 KHz/50 hrs		
Impacts, g		300		

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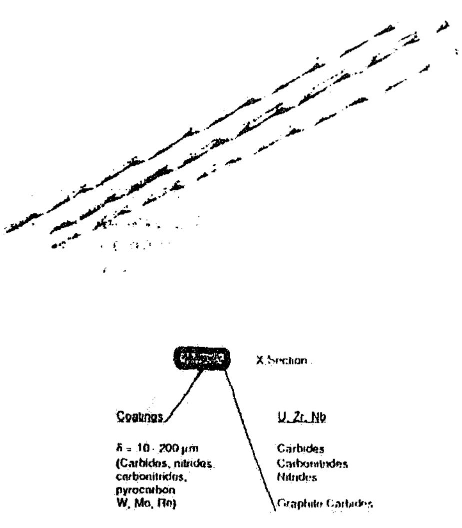
Russians have developed several families of high temperature nuclear fuel and have tested them in many ways in material laboratories, nuclear reactors, and in hot cells for post-irradiation properties. Carbide fuels have demonstrated the highest temperature capabilities with good material properties. U, Zr, Nb tricarbide alloys are selected for NTRE fuel, the exact composition depending on core location. Low uranium concentration is used in the highest temperature, aft fuel bundles, and in the center of the core to flatten the radial power distribution profile. The favored geometry of individual fuel elements is that of thin, twisted strips. Coatings of carbides, nitrides, carbonitrides and pyrocarbon mixtures of tungsten, molybdenum, and rhenium have been developed and tested for corrosion resistance and thermal strength characteristics.

Fuel Element Rods Have Been Tested Thoroughly

Rod Fuel Element



Test Conditions

Parameter			Before Reactor	In Reactor	Post-Reactor
Strength	Tension	Temperature, K	2100	T ≤ 3100, τ ≈ 4100s	10^{20} n/cm², T ≈ 300K
	Bending		<3500		
	Torsion		2100		
Thermal Technology	Heat Conductivity		2300		
	Heat Capacity		2300		
	Thermal Expansion		2800		
	Plastic Deformation		<3500		
Evaporation of Material		Vacuum, He, H_2	≤ 3200		
Melting			3900		
Radiation Swelling				T ≤ 3200 (He, H_2, vac)	T ≤ 2500K
Output of Fission Products				T ≤ 3200	
Interaction with H_2, N_2 Additives of O_2, H_2O, CH_4			T ≤ 3500, P ≤ 20MPa, τ = 800s	T ≤ 3100, P ≤ 10MPa, τ =	
Surface erosion neutrons cm2			T ≤ 3000, ρu^2 - 2 10^5		
Heat Generation kW cm³			90	30	

Operation Conditions

	Graphite Carbides	Carbo nitrides	Carbides	
T, K	<2700	<3100	<3500	
q_v kW/cm³	<80	<30	<40	
τ, s	10^4	15000	$5x10^3$	>7000
Coolant	H_2	N_2, H_2 + δN_2	H_2	H_2 + δC

Coatings reduce relative fission products release by 30 - 200 times

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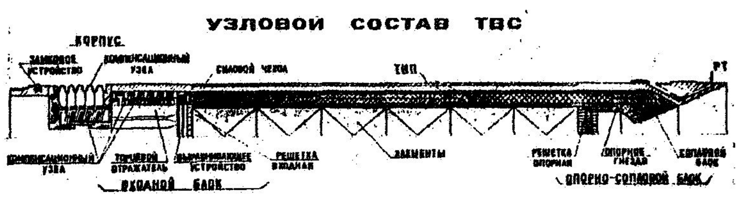
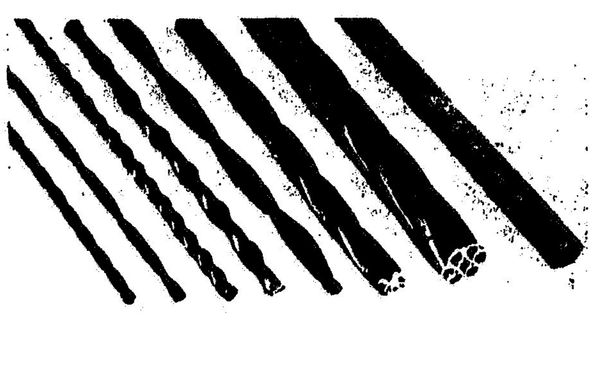
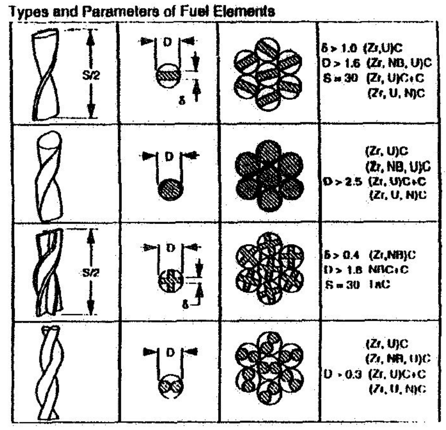
Engeropool fuel assemblies for NTRE are about a meter long and 30 to 45 mm in diameter. The forward, inlet end runs cooler than the rest of the assembly, and it contains an adjustable flow metering orifice, neutron reflector, and thermal expansion compliance (bellows). The center of the fuel assembly contains several bundles of fuel elements placed in series, flow passing through each sequentially, so that the hottest gas exits the aft end of the last bundle. A grid plate holds the fuel element bundles in place and allows the hot gas to pass through to the outlet plenum of the fuel assembly. Each grid plate resembles fuel bundles, except that the carbide elements are fused together and contain no uranium. The outlet plenum delivers gas to either a DeLaval rocket nozzle or to a subsonic diffuser (selected for our Mars engines). The outer envelope is metallic and hydrogen cooled within the reactor, and it is insulated internally with several layers of graphitic material.

Each Fuel Assembly Is a Fully Integrated Unit





Types and Parameters of Fuel Elements



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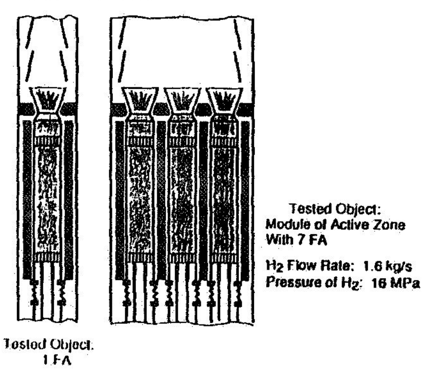
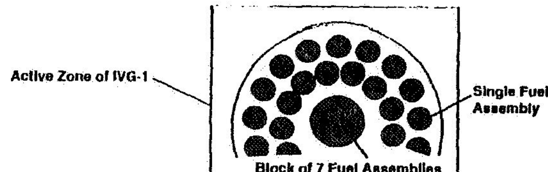
Fuel assemblies have been tested in the steady state nuclear reactor, IVG-1, both singly and in clusters of seven in hydrogen, and under high temperature and pressure at thermal neutron fluxes above 10^{15} neutrons/cm^2-sec.

Typical Test Arrangements of Fuel Assemblies in IVG-1 Reactor

Obtained Conditions of Tests

1. Power of 1 FA $\leq$ 11 MW
2. Temperature of $H_2 \approx$ 3100K
3. Power Density q_v^{max} 35. MW
4. Heat Flux q_s^{max} 13 MW/m
5. H_2 Temperature Transient 150°K/s
6. Reactor Starts Per Month, 2
7. Thermal Neutron Flux $\approx 2 \cdot 10^{15}$ neutrons/cm^2 -s





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Many reactor materials have been fabricated and tested by Energopool in laboratories, in reactors, and in post-irradiation hot cells. Structural, neutron moderating, neutron reflecting, and neutron absorbing materials have been tested to high fluxes, fluences, temperatures, and immersion times in hydrogen and other media.

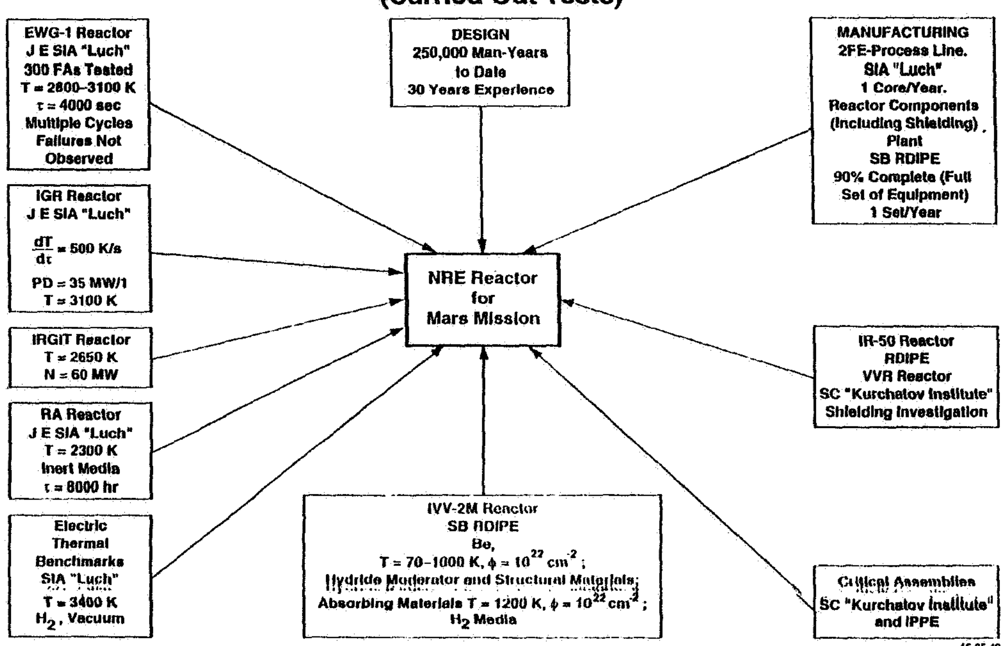
Materials Testing Results

Materials	Test Parameters: Neutrons cm^2/s	Neutrons cm^2	T, °K	Life-Time, Hours	Medium	Notes
Steel and its Alloys (18 Types)	10^{13} 10^{14}	up to $3 \cdot 10^{20}$	77 ... 1100	500 - 12,000	H_2, He Vacuum	
Be, Be-Al, Be with Coatings	10^{12}... 10^{14}	up to $2 \cdot 10^{21}$	77 ... 1200	500 - 2,400	H_2, He H_2O Vacuum	During Some Experiments T = 20°K q_v = 150 W/cm3
ZrH, ZrH + B, LiH	10^{12}... 10^{14}	$4 \cdot 10^{19}$ up to $2 \cdot 10^{21}$	400 .. 1,000	500 - 12,000	H_2 CO_2 He	
Absorbing Elements	10^{13}... 10^{14}	up to $2 \cdot 10^{21}$	600 .. 1,300	500 - 10,000	H_2, He Vacuum	q_v^{max} = 700 W/cm3

The experimental capability of Engeropool is rather complete. Between six companies they have experienced analysis, design, manufacturing, and test personnel. Energopool has facilities that cover the range of critical assemblies, shielding investigation, material and equipment manufacturing, reactor safety analysis, and many testing laboratories. These laboratories include facilities for material investigation in pre, in-pile, and post-irradiation environments, hot cells, and a variety of research reactors for fuels investigation.

Experimental Capabilities of CIS Test Facilities for Validation of NRE Reactor Development

(Carried Out Tests)



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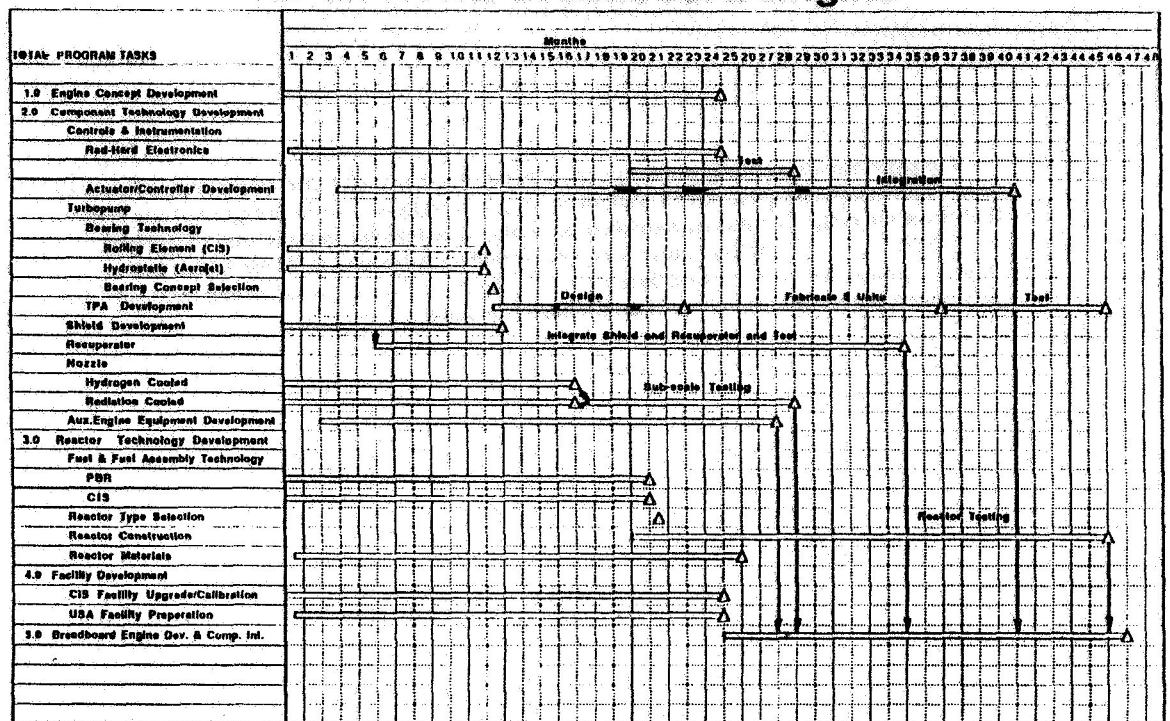
Parallel 4-Year Technology Development for an NTRE Breadboard Engine



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Tuesday, October 20, 1992

17 69 5c/rt/81

Summary

Mel Bulman

GenCorp Aerojet • Energopool • Babcock & Wilcox

17 89 5n/u54

Our Integrated Engines Provide

Safety and Reliability

- Simple Thermodynamic Cycle
- Integrated Auxiliaries Simplify Propulsion
  - Start System
  - RCS (No Igniters, O_2, Combustion)
  - Electric Power and H_2 Refrigeration
  - Four Core Cooling Systems
- Improved Engine Start (Preheat)
  - No Thermal Shocks
  - Enhances Multiple Engine Safety
  - No LH_2 in Core (Reduced Reactivity Insertion)
  - Thermal and Acoustic Damping
  - Assured Restart
- High Margins – Long Life
  - Low Fuel Temp and Stress (4600°F)
  - Low Turbine Temperature (400°F)
  - Low Nozzle Temperature (600°F)
  - Low Moderator Temperature (400°F)
  - No Deep Thermal Cycles

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Our Integrated Engines Provide

Mission Benefit

- **Improved Mission Average Isp**
  - Heterogeneous Reactor
  - Greatly Reduced After Cool Loss, Save > 100 K lb LH_2
  - LH_2 Refrigeration Option
  - OMS Thrust at > 700 sec Isp (w/o Pump Start)
  - ACS Thrust at > 500 sec Isp
- **Improved Engine Thrust/Weight**
  - High Power Density Reactor
  - High Pc (Reduces Shield and Nozzle Size and Weight)
- **Operational Benefits**
  - Deep Throttling (Enables Multiple Burn TMI)
  - ~ 100 kWe Electric Power/Engine
  - Rapid Restart

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Our Integrated Engines Provide

Low Life Cycle Cost

- **TRL 4 to 6 for Major Components**
  - CIS Fuel Developed
- **Smaller, Lower Cost Components**
  - High Pc
  - Nozzle
  - Pressure Vessel
  - Shield
- **High Pc Enables Small ETF**
  - High Pressure Storage
  - High ΔP Scrubbers
- **Reduced ETO Cost**
  - Reduced IMLEO
  - Smaller Payload Bay
- **Design Flexibility and Growth Potential Reduces Cost**
  - Recuperated Cycle
  - Electrical Power System

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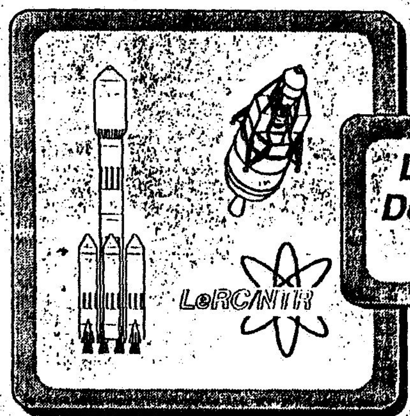

Lunar NTR Vehicle Design & Operations Study

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JH921007 00A

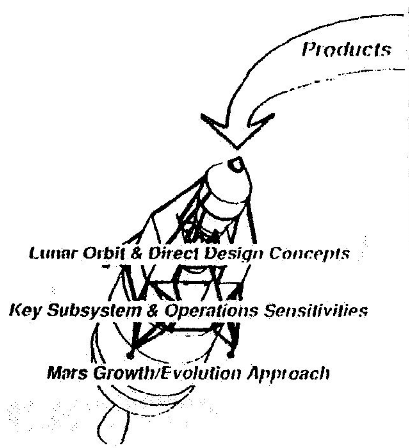
LTS/NTR Design & Operations Study

Objectives

Perform an Evaluation of the Potential Applications of a Specific Nuclear Thermal Rocket (NTR) Design to Past and Current (First Lunar Outpost) Mission Profile(s) for Piloted and Cargo Lunar Missions, and to Assess the Applicability of Utilizing Lunar Vehicle Design Concepts for Mars Missions

- Assess NTR Propulsion For Lunar Vehicle Concepts Based on Existing Mission Profiles
- Define and Size the Stage/Transfer Vehicles for Lunar NTR Applications
- Perform an Operational and Programmatic Assessment
- Perform a Chemical/NTR Lunar Concept Comparison

Products



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Reqts Provide Basis for NTR Vehicles

Mission Statement

- Support Exploration & Habitation of Lunar Surface with consideration for evolvability to Mars
- Lunar IOC 2000-2005
- Mars IOC: 2005 Cargo 2007 Piloted

Functional Reqts	System Reqts: Lunar Direct	System Reqts: Lunar Orbit Rendezvous
1.0 Prelaunch Proc		
2.0 Launch Ops	NLS & Saturn V derived launch systems	66t & 135 t launch system Only 2 HLLV Flts per mission 2nd HLLV Flt within 3 days
3.0 Transfer to Lunar Surface	4 Day Transit Cargo Mission- 33 t Manned Mission-5 t LOI Ops Altitude = 300 km Post TLI Disposal ΔV = 30 m/s Post LOI Disposal ΔV = 860 m/s	3 Day Transit LOI Ops Altitude = 300 km Post TLI Disposal ΔV = 30 m/s Post LOI Disposal ΔV = 860 m/s Deliver at least 5 t cargo & crew of 4 on piloted mission
4.0 Surface Ops	Surface Stay = 45 Days	Surface Stay = 180 Days
5.0 Earth Return	NTR reuse altitude = 500 km Return 200 kg to earth Post TEI disposal ΔV = 194 m/s	NTR reuse altitude = 500 km Return 500 kg to earth Post TEI disposal ΔV = 194 m/s

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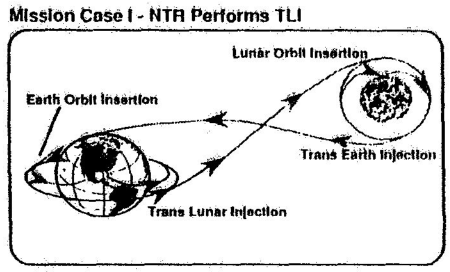
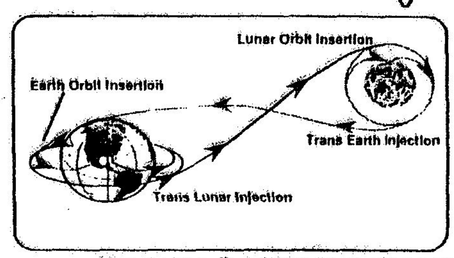
LR920812-System Reql

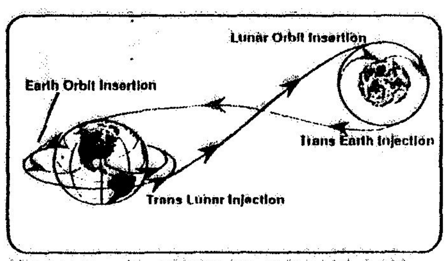
Lunar Mission Options

Mission Case I - NTR Performs TLI



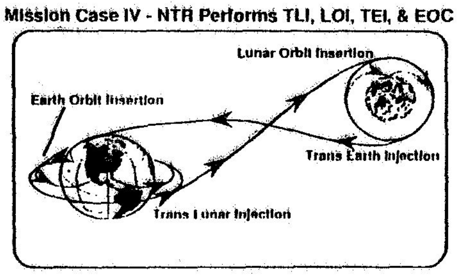


Mission Case II - NTR Performs TLI & LOI



Mission Case III - NTR Performs TLI, LOI, & TEI

Mission Case IV - NTR Performs TLI, LOI, TEI, & EOC



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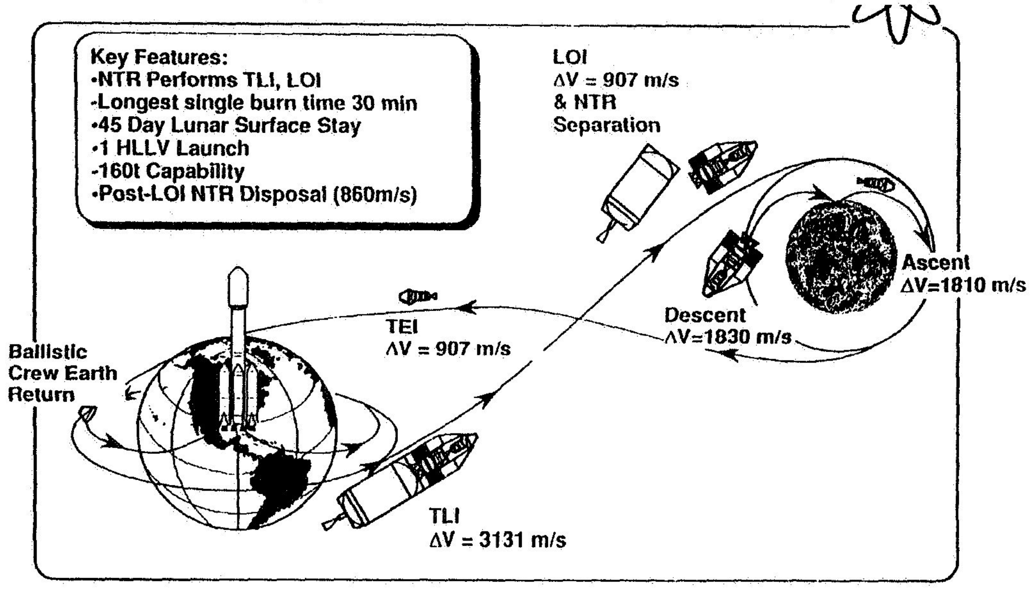
LD Space Operations Overview



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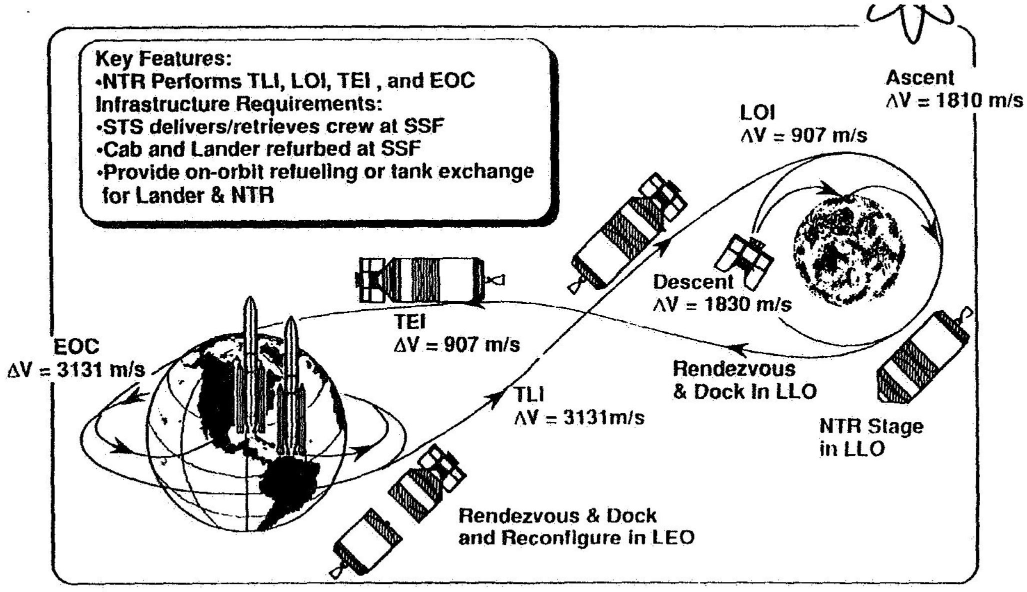
LR921009-LD Ops

LOR Space Based Operations Overview



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LR921009 LOR Reuse Ops

System Design Considerations

- Expendable vs. Reusable
  - Operations Complexity Too High For Reusability
  - Performance Maximization Achieved With Expendable Mission
  - Safety Concerns Lessened With Expendable Mission
- Shielding Considerations
  - NERVA Disk Shield
  - Modified Disk Shield Optimizes Design and Use of Propellant
  - Propellant and Tankage
  - Lander Propellant and Structure
- Launch Vehicle Considerations
  - Lunar Direct Mission
    - Smallest Launch Vehicle Necessary to Complete FLO Mission
  - Lunar Orbit Rendezvous
    - Complete Reasonable Lunar Architecture Using Medium Sized Launch Vehicle
- Thermal Protection Considerations
  - Active Refrigeration Too Heavy For Benefit & Abort Mission If Failed MLI & SOFI
- Lander Considerations
  - Lunar Direct Mission
    - 2 Stage Cryo/Storable Removes Long Term Hydrogen Storage on Orbit
  - Lunar Orbit Rendezvous
    - 1 1/2 Stage Cryo/Cryo Out-performs 2 Stage Lander Consistently in Past STV Studies
- Material and Construction Considerations
  - Aluminum Lithium Technology On Schedule For Flight Use By 2005
  - Isogrid Construction Promising For Structural Considerations
- Engine Configuration
  - Single vs Cluster
  - 25k vs 50k vs 75k

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JH921012 11A

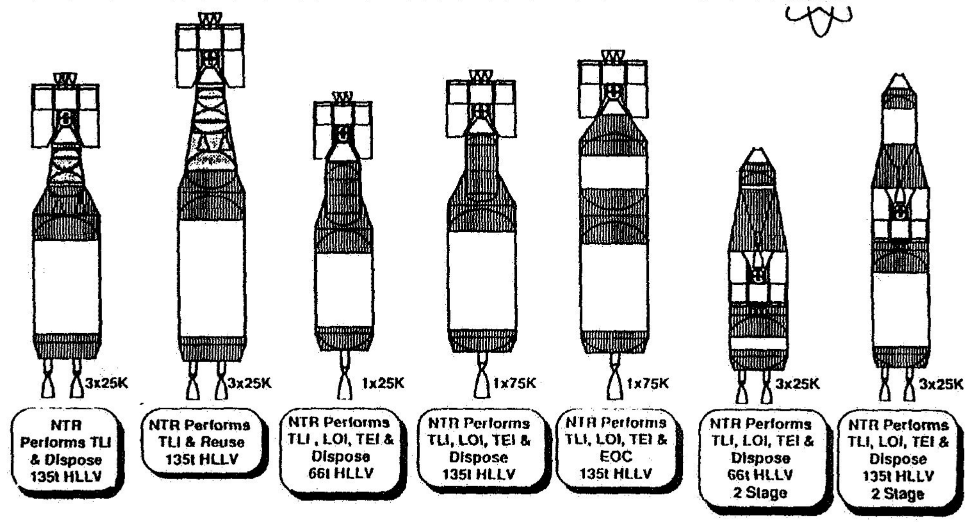
Lunar Orbit Rendezvous Configurations

The chart below shows the top seven candidates of the Lunar Orbit Rendezvous configurations. We started with eighteen possible configurations in this catagory, and through performance runs, design constraints and operational issues, that eighteen was narrowed to the following seven. The missions that utilized a cryogenic liquid oxygen and liquid hydrogen TEI stage were extremely close in performance to those missions using NTR to perform the TEI burn. Therefore, it was beneficial to show the elimination of an entire technology, use of a LOX/LH_2 stage, and to show that a lunar mission can be supported solely by NTR technology. Another criteria that eliminated candidates was performance: at least 5.00 tonnes to the lunar surface on a piloted mission. Also eliminated in the earlier phases of the study were two HLLV candidates. We started with four HLLV candidates: 66t, 105t, 132t, and 135t launch capacities. We narrowed that field to two candidates based on past STV analysis showing the 132t and 135t vehicles virtually even on performance. Of the three that were left, 66t, 105t, 135t, we eliminated the 105t because of study complexity and to demonstrate that NTR can be utilized on the two extreme launch vehicles and, therefore, everything in between.

Three of the configurations shown below started with a cluster of three 25Klbf NTR engines, but with further analysis their performance was greatly enhanced by going to a single engine configuration. Also, two of the configurations are two-stage NTR configurations. The first NTR stage performs the TLI burn and is then staged off to perform a lunar assist disposal burn into heliocentric space. The second NTR stage then performs the rest of the mission and is also disposed of after the TEI burn.

JH920803 LPLOR Configs

Lunar Orbit Rendezvous Configurations



Stack Mass	254.75	263.25	128.56	265.45	267.83	131.73	261.68
Perfomance							
Piloted	26.55	19.55	7.21	38.68	19.35	5.88	38.53
Cargo	47.55	39.62	25.66	61.52	44.93	23.69	60.39

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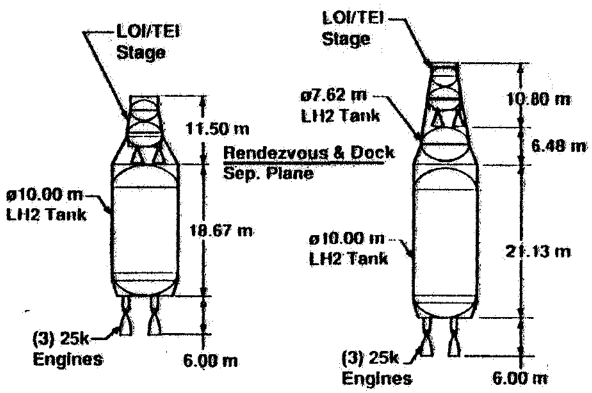
JH921013-03A

Lunar Orbit Rendezvous Configurations

NTR Performs TLI and Disposed
NTR Performs TLI and Reused



LOI/TEI Stage

Component	TLI & Dispose t	TLI & Reuse t
Structure	2.46	2.39
Tankage	0.99	0.93
Subsystems	2.83	2.83
Engine Structure	0.51	0.51
Engines	0.80	0.80
Contingency (15%)	1.14	1.12
Total Dry	8.73	8.58
Propellant	41.18	35.71
Total Wet	49.91	44.29

NTR Stage

Component	TLI & Dispose t	TLI & Reuse t
Structure	0.87	1.83
Tankage	5.10	7.20
Subsystems	2.11	2.64
Engine Structure	0.75	0.86
Engines	11.18	11.18
Shield	0.00	0.00
Contingency (15%)	3.00	3.56
Total Dry	23.01	27.27
Propellant	91.88	116.13
Total Wet	114.89	143.40

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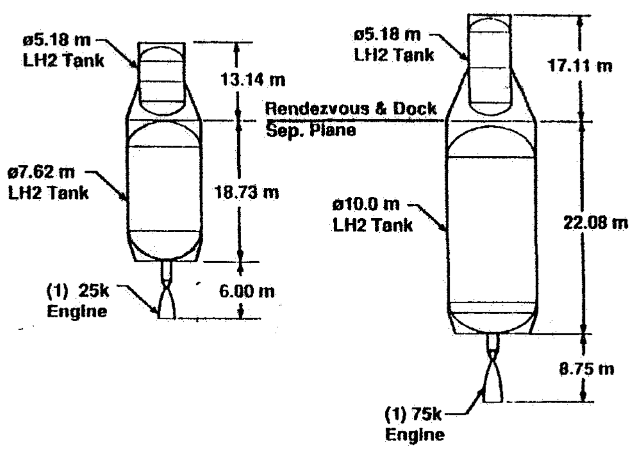
JH921013 05A

Lunar Orbit Rendezvous Configurations

NTR Performs TLI, LOI, TEI and Disposed - 66t HLLV & 135t HLLV



NTR Stage

Component	66t t	135t t
Structure	1.37	1.95
Tankage	3.71	6.80
Subsystems	1.98	2.87
Engine Structure	0.41	0.96
Engine	3.73	6.83
Shield	1.50	4.50
Contingency (15%)	1.91	3.59
Total Dry	14.61	27.50
Propellant	63.73	123.09
Total Wet	78.34	150.59

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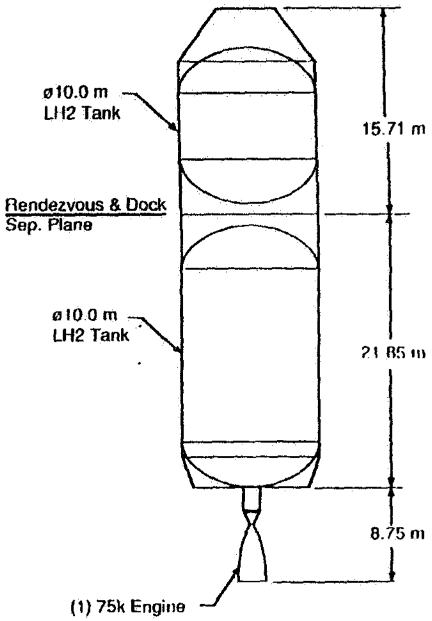
Lunar Orbit Rendezvous Configurations

NTR Performs TLI, LOI, TEI and EOC - 135t HLLV



NTR Stage

Component	t
Structure	2.91
Tankage	9.70
Subsystems	3.45
Engine Structure	0.96
Engine	6.83
Shield	4.50
Contingency (15%)	4.25
Total Dry	32.60
Propellant	160.08
Total Wet	192.68

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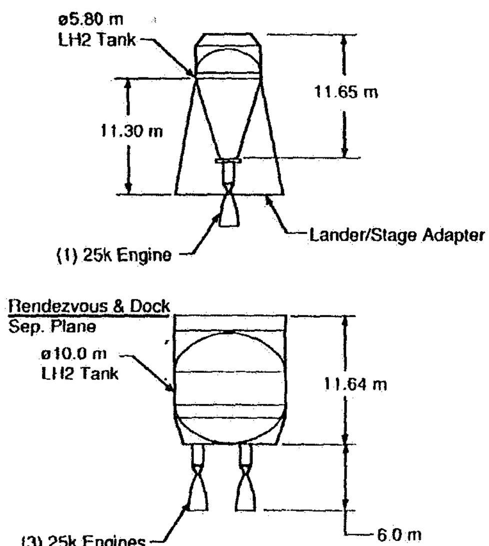
JH921013 07A

Lunar Orbit Rendezvous Configurations

2 Stage NTR Performs TLI, LOI, TEI and Dispose - 66t HLLV



NTR Stage 1 & 2

Component	t
Sturcture	2.59
Tankage	4.50
Subsystems	1.69
Engine Structure	0.76
Engine	14.91
Shield	1.50
Contingency (15%)	3.89
Total Dry	29.84
Propellant	54.42
Total Wet	84.26

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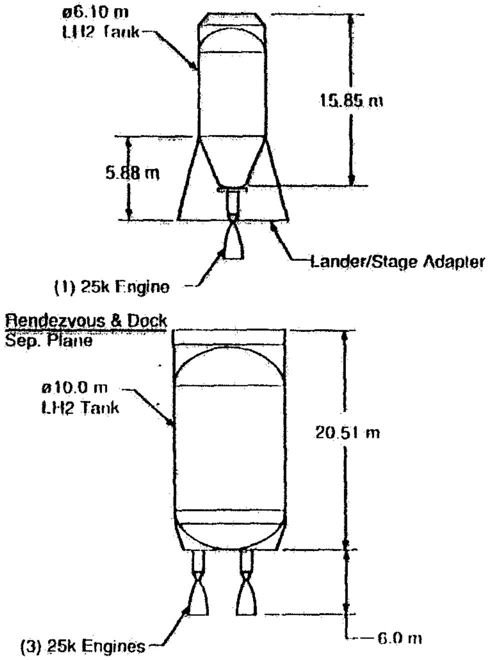
Lunar Orbit Rendezvous Configurations

LeRC/NTR

2 Stage NTR Performs TLI, LOI, TEI and Dispose - 135t HLLV



NTR Stage 1 & 2

Component	t
Structure	1.79
Tankage	6.60
Subsystems	2.73
Engine Structure	0.81
Engine	14.91
Shield	1.50
Contingency (15%)	4.25
Total Dry	32.59
Propellant	114.55
Total Wet	147.14

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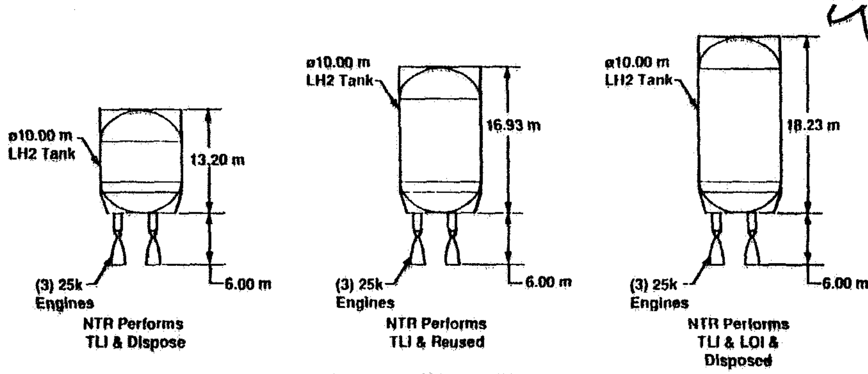
Preliminary LD NTR/TLI Configurations

LeRC/NTR



NTR Stage

Component	TLI & Dispose t	TLI & Reuse t	TLI, LOI & Dispose t
Lander/TLI Adapter	1.49	1.49	1.49
Tankage	4.40	5.20	5.50
Subsystems	.81	.93	.97
Engine Structure	2.01	2.29	2.38
Engines	11.18	11.18	11.18
Shield	0.00	0.00	0.00
Contingency (20%)	3.98	4.22	4.30
Total Dry	23.87	25.31	25.82
Propellant	59.60	80.64	87.96
Total Wet	83.47	105.95	113.78

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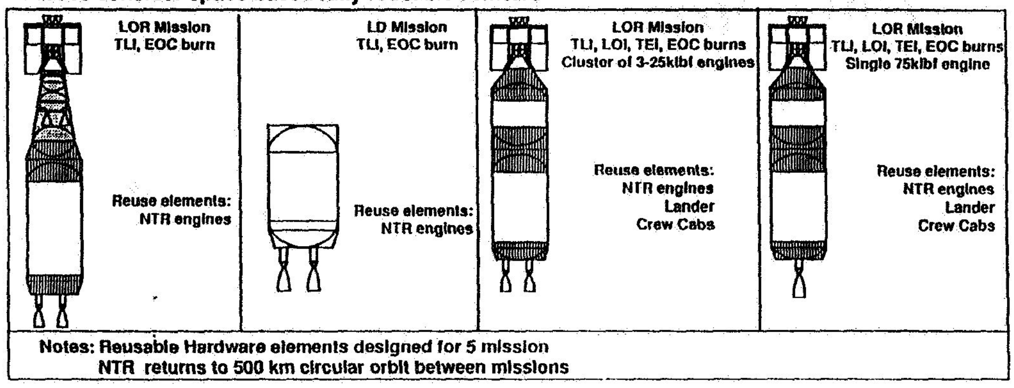
NTR Reuse Examined in Study

•Developed vehicle mass properties, payload capabilities, space operations
—2 cases consider reuse for NTR only
—2 cases consider space based fully reusable vehicles



•Identified infrastructure needs (assumed existing)*
—STS for crew delivery/retrieval
—SSF for refurb
—Capability for on-orbit refueling or tank exchange

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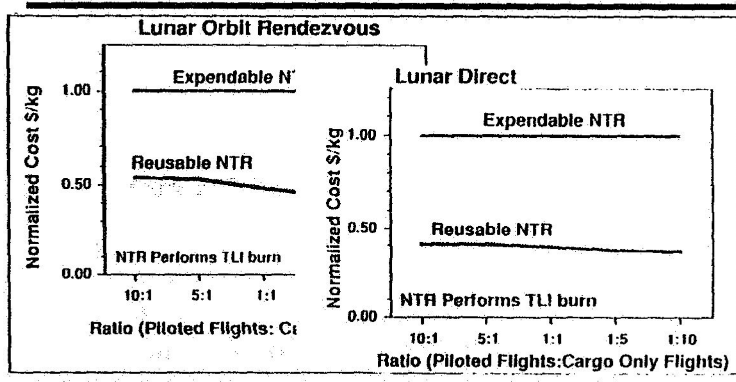
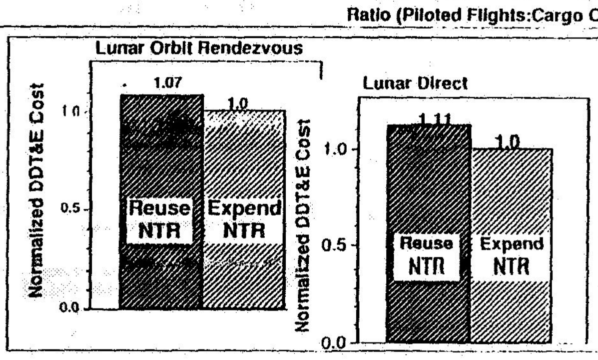
*Infrastructure costs (elements & associated operations) were not included in cost analysis

LR920819-REUSE GRD RULES?

Reuse Cost Analysis



Reuse Reduces Vehicle Recurring Cost approximately 50%



Reuse Increases Vehicle DDT&E approximately 10%

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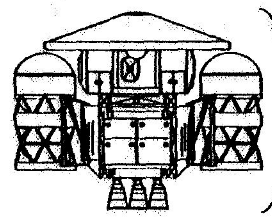
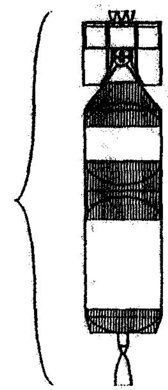
LR920818 Reuse$

NTR/Cryo Reusable System Comparison

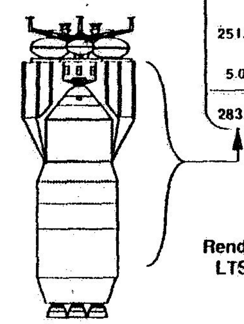


Space Based 90-Day Study Derived LTS

Reusable Rendezvous & Docked NTR/Cryo LTS



		MASS PROPERTIES ("t")	
27.58	27.58	Dry Mass	42.90
		(NTR/TLI Stage)	(32.60)
		(Lander)	(12.30)
251.0	174.0	Total Propellant	185.0
5.0	15.26	Piloted Cargo	14.8
283.6	217.50	TOTAL SYSTEM	242.78



Rendezvous & Docked LTS (2 135 t HLLVs)

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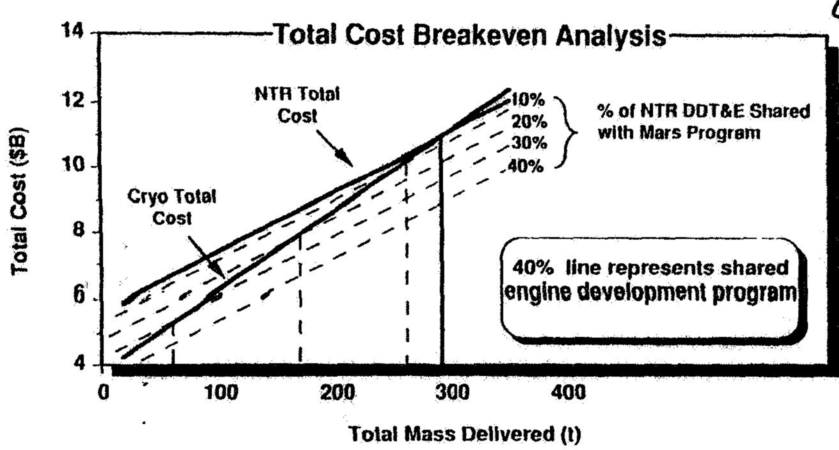
JIB921016 10A

Mars Evolution Key to Affordability



Sharing Development Cost of Common Elements with Mars Program Lowers Total Cost of Lunar Missions

Example: Splitting the engine development cost between the lunar and Mars programs shifts breakeven point from 300 t to less than 60t

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IB921012 Mars evolution

Conclusions

- **Near term NTR Provides Feasible Alternative to Cryo system for Lunar Missions**

-Performance
NTR LD concept offers smaller IMLEO for same payload capability as cryo system
NTR LOR offer greater payload delivery capability for same IMLEO
-Cost
NTR more cost efficient ($/kg) than cryo system
NTR Development cost greater than cryo systems
-Ops
LOR option requires on-orbit cryo transfer (technology risk)

-Schedule
1st cargo launch capability in 2002

- **Near term NTR enable efficient evolution to Mars**

-Cost
Shared development cost of common elements enhances affordability
Hardware commonality
-Mission
Lunar NTR adaptable to wide range of mission architectures (Direct, MOR)

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LR921009-Conclusion

N93-26923

Mission Design Considerations for Nuclear Risk Mitigation

Mike Stancati
&
John Collins

• Safe Return of NTR to Earth Orbit	Pulsed cooldown propellant can be used to lower capture orbit to selected operations altitude
• Lunar/Mars NTR Disposal	Modest cost, low risk disposal to heliocentric orbits for all transfer trajectories

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Aim Point Bias Offsets Targeting Errors

One of the operational safety concerns for nuclear propulsion is how to manage safe return of a nuclear powered transfer vehicle to Earth orbit. Although the mission profiles used in this study call for crew return by an Earth Crew Capture Vehicle (ECCV), capturing the transfer vehicle offers the added flexibility and possible cost savings of reuse. The return orbit must be low enough to be accessible from Earth launch at reasonable cost, yet high enough to ensure safety.

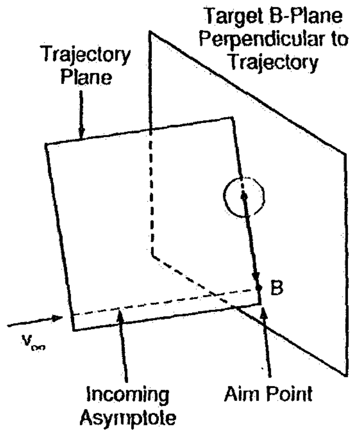
Experience shows that orbit insertion errors are caused by some or all of the following factors:

- Errors in Trajectory Correction Maneuvers (TCM), resulting from off-nominal thrust level, direction, or duration. These may be caused by the propulsion or attitude control subsystems.
- Inherent uncertainty in determining the spacecraft's orbit
- Small errors in precise location of natural bodies

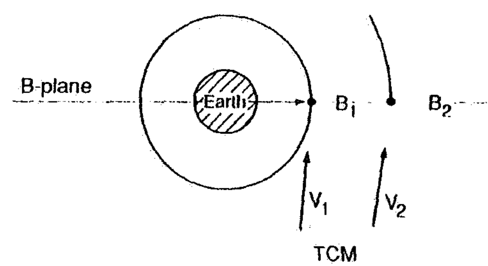
However, the resulting variations in spacecraft orbit parameters are small; orbit insertion altitudes vary by only a few kilometers. The performance of a nuclear thermal rocket should be similar to past experience with chemical systems. The critical item, then, is to select a nominal return orbit that matches lifetime characteristics with the needs of short-term storage in Earth orbit.

The aim point can be biased so as to raise the distance of closest approach, and capture into some orbit higher than the desired one. After exact position and status of the vehicle is determined, a series of smaller burns lowers the orbit to match the final size. This approach will be the basis of a proposed strategy for making effective use of pulse-mode cooldown propellant.

Aim Point Bias Offsets Targeting Errors



- Targeting errors result from off nominal TCM burns, uncertainty in orbit determination
- Typical injection errors are small: altitude dispersions of a few kilometers
- Damp out dispersion effects by changing the aim point to a higher orbit (elliptic or circular), then use small impulses to lower to desired final orbit



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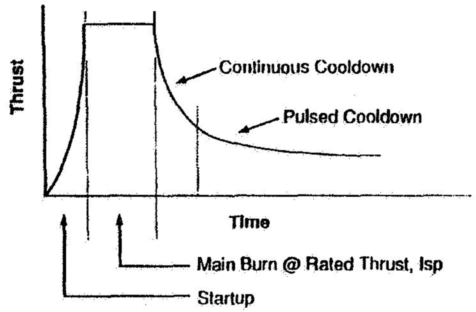
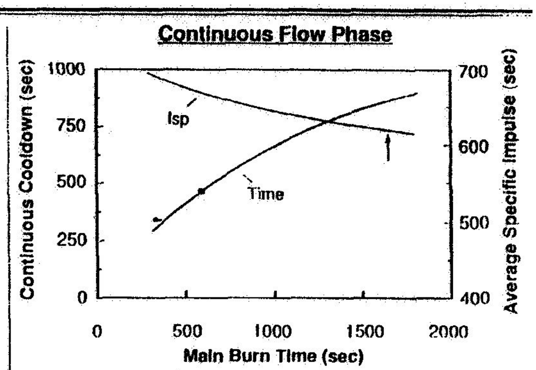
Cooldown Propellant Characteristics

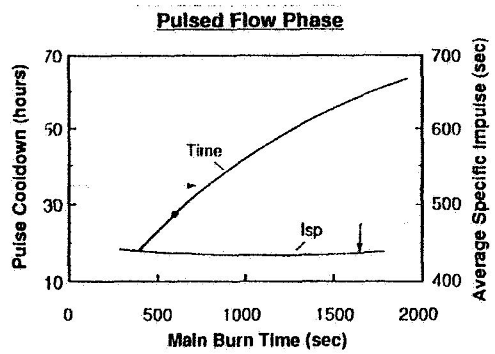
Since NTP systems must cool the reactor with flowing hydrogen after every main burn, it would be desirable to use as much as possible of this propellant for productive thrust. The continuous cooldown flow lasts for a few minutes, and is assumed to handle part of the required capture impulse. The pulsed flow lasts over several hours, with the exact profile depending on the main burn duration. Pulsed flow averages a specific impulse of about 440 seconds, but at a very low thrust level.

The table in the lower right corner opposite shows the four phases for a main burn of 600 seconds, typical of Earth orbit capture burns for return from the Moon. In this case, the pulsed flow must occur over 31.5 hours to keep the reactor within the specified temperature range.

Cooldown Propellant Characteristics







Phase	Δ Time	Propellant (kg)	Average Thrust (lbf)	Average Isp (sec)
Startup	60 s	3,127	17,411	737
Main	600 s	24,737	75,000	825
Continuous	458 s	1,748	5,635	670
Pulse	31.5 h	1,483	12.7	438

Source: Flight Engine Program (FEP) Runs by Aerojet, Aug. 1970

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Altitude Profile During Pulsed Cooldown

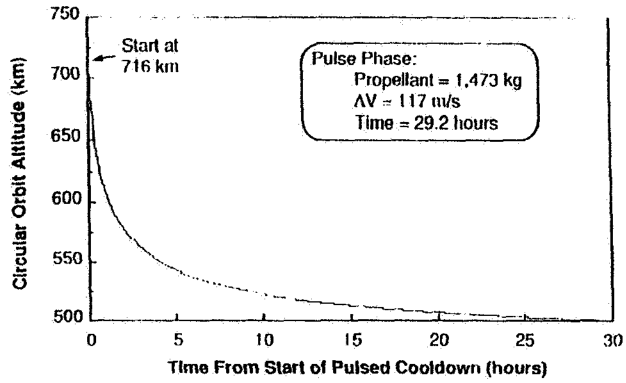
The baseline profile for a piloted mission to either the Moon or Mars is to separate the crew in an ECCV for direct reentry at Earth return. This eliminates the operational concerns of crew safety on board during the extended cooldown time interval. Modeling of the pulsed flow shows a total propellant requirement of 1,473 kg for cooldown purposes. From the LTV mass requirements, shown elsewhere in this study, this translates to a ΔV of 117 m/s, assuming the maximum effective thrust could be imparted by the propellant flow.

The problem is to match the required cooldown propellant flow with an orbit modification strategy that meets mission requirements, while deriving maximum value from the available ΔV. The approach used here is to accept the pulsed flow profile as given in the Aerojet FEP output runs referenced on the previous page (although it may be possible to modify it), and to use each low thrust impulse to lower the altitude of the initial capture orbit. We begin with a circular orbit at some altitude, and proceed to apply a sequence of many small impulses to lower to the desired 500 km orbit at the end of pulsed cooldown. Starting with a circular orbit at 716 km will produce the desired final altitude of 500 km at the end of this thrusting program.

This use represents one example of how the pulsed cooldown propellant may be used for transfer vehicle thrust. A complete characterization for the range of engine sizes considered in this study will require burn simulations for various thrust levels, burn times, and numbers of engines staged together.

Altitude Profile During Pulsed Cooldown

- LOR Mode, with NTR performing 4 main burns
- Crew separates from LTV on approach and returns in ECCV
- 600 sec burn for Earth capture



- Efficient use of pulse-mode cooling for thrust is the active constraint in selecting an initial capture orbit
- Biasing to counter possible orbit insertion errors is easily satisfied by this initial selection

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NTR Disposal for Lunar Missions

Two broad categories of long-term disposal orbits have been examined for use with nuclear thermal propulsion in lunar and Mars transfer vehicles. These are: Earth orbits at altitudes high enough to ensure long life before reentry, and various heliocentric options. Although all of these offer real possibilities for reactor disposal, selection may depend on programmatic guidelines. As a conservative approach, we consider the heliocentric options to be preferable, so long as the propulsion requirements are reasonable. As the table opposite shows, ΔV to reach a particular disposal orbit is highly dependent on where the transfer vehicle is when the disposal operation commences. Since this study considers a variety of NTP use scenarios, there is no single lowest-cost solution.

NTR Disposal for Lunar Missions

- Want permanent solution:
  - small long-term risk of Earth reentry
  - no requirement for active management

- Two classes of candidates: Earth orbits and heliocentric orbits

- Cost (ΔV) influenced by location when disposal sequence begins

- Final selection will depend on program's balancing of risk (real and perceived), and cost

Disposal Starts From	NTR Final Disposal Location			
	Earth Orbit ≥ 1000 km	Super-GEO	Earth-Crossing Heliocentric	Stable Heliocentric
Post-TLI	3110	2166	194	1450
Lunar Orbit	4210	1758	860	2550
Post-TEI	3110	2166	194	1450
LEO	200+	3859	3267	4550

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Crossing Orbits

Previous work by SAIC identified a class of Earth-crossing orbits that lie just inside or just outside Earth's orbit, and that are slightly out of the ecliptic. Using a Monte Carlo simulation code to estimate the lifetimes and reentry probabilities of a body in such an orbit shows promising statistics for use as a disposal location. The Earth reencounter probability shown in the graph represents a slightly higher risk than collision with an asteroid in similar time periods.

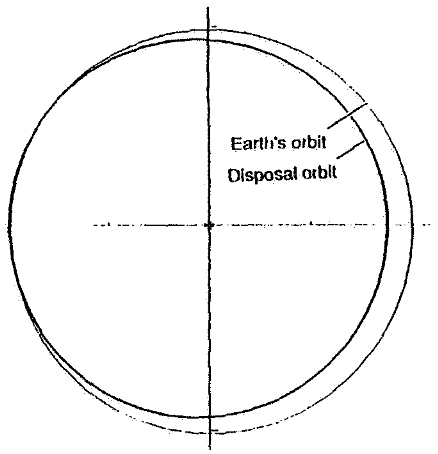
As the table on the previous page showed, the crossing orbits are easily attained from most points in lunar mission profiles. The exception is disposal of a vehicle from LEO.

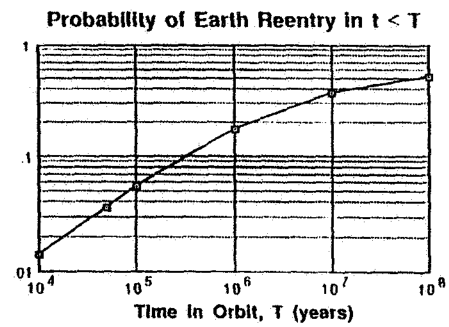
Crossing Orbits

- Graze Earth's orbit at 0.88 x 1.0 A.U. or 1.0 x 1.15 A.U.

- Are slightly inclined to minimize reentry probability: i > 2 deg

- Are predicted to have: **[Planetary Encounter Probability Analysis (PEPA) code]**
  - mean lifetimes of the order of 10^7 years
  - probability of Earth reencounter in ≤ 10,000 years of 1-2%
  - probability of Earth reencounter in ≤ 1,000,000 years of 17-18%





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Stable Orbits

The second category of heliocentric orbits was also identified by SAIC as a possible permanent storage location for hazardous waste in space.[2] This analysis was one part of a large effort to explore space-based alternatives for nuclear waste disposal conducted during 1977-79. These orbits are of interest because they are predicted to endure for a very long time without becoming planet-crossing orbits. Two bands of these stable orbits have been identified, as shown opposite. The one of most interest for Earth-Moon and Earth-Mars cases is at 1.19 A.U., between Earth and Mars.

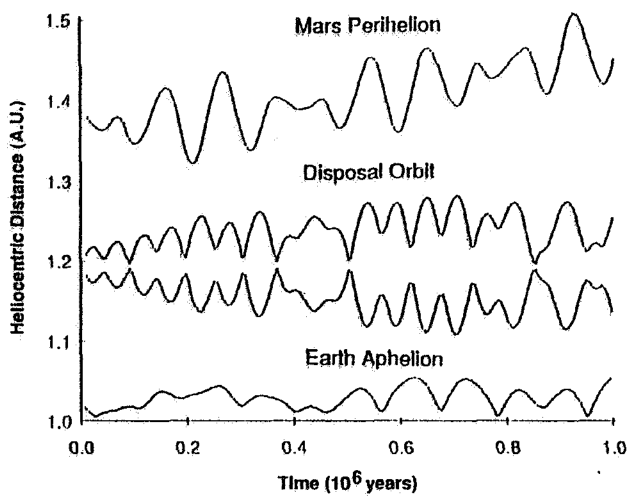
The orbit starts out circular, but becomes elliptic "quickly" in the long view of the situation, as shown on the graph in the lower left corner. This graph plots heliocentric distance as a function of time (note the x-axis scale!) for the periapse and apoapse of the stable orbit. The Mars peripase and Earth's apoapse are also plotted. All four show significant variations over the one million year time frame, but the stable orbit never crosses its closest planetary neighbors' paths. This means that, with no further active management, placing an object in the stable orbit is sufficient to remove the real risk of the on-board radiation hazard.

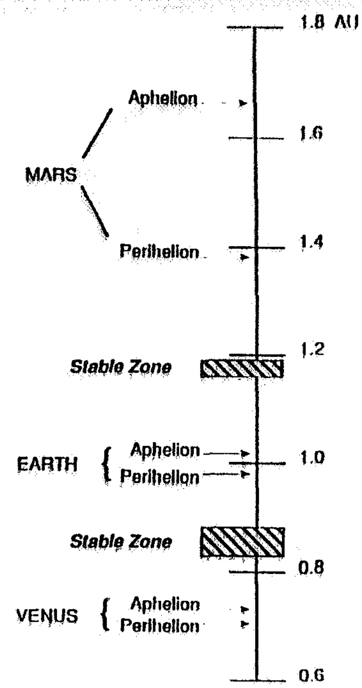
As the earlier table indicated, significant impulses (1450-4550 m/s) are required from the lunar flight path to deliver the LTV to this orbit. Although there is a cost difference over the crossing orbit, the stable orbit offers a greater risk reduction potential. Whether the additional risk reduction is required will depend in large measure on program guidelines and policy.

2 Friedlander, A. L. and D. R. Davis, "Long-Term Risk Analysis Associated With Nuclear Waste Disposal in Space," SAIC Report No. 1-120-062-112, prepared under contract NAS8-33022 for NASA/MSFC, December 1978.

Stable Orbits

- An orbit is stable over time T if a body in that orbit doesn't cross a planet's path in T
- Starts at 1.19 x 1.19 A.U., becomes elliptic, but doesn't cross Mars or Earth





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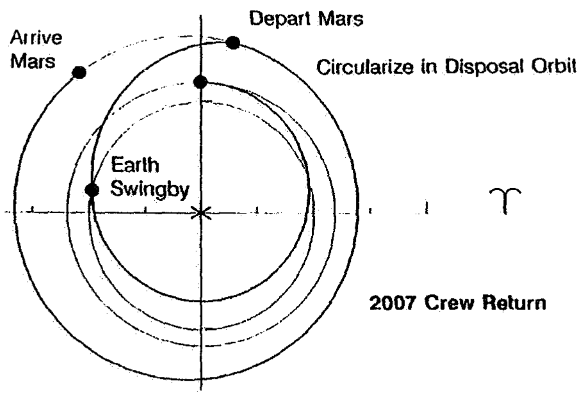
NTR Disposal for Mars Missions

Two options are considered for Mars missions: the stable heliocentric orbit, or disposal along the interplanetary trajectory that the transfer vehicle is following. For the former, the ΔV requirements for two split/conjunction mission pairs are shown on the facing page. Cargo missions need two impulses to leave Mars and to circularize. Crew mission trajectories are modified to perform Earth gravity assist after ECCV separation, saving roughly 2 km/s impulse. The orbit plot show the 2007 crew return profile, with Earth swingby to final capture burn at 1.19 A.U.

The second option is to leave the transfer vehicle in its flight path. In all cases, the flight path crosses at least one planet's path, setting up possible unintended gravity assists in the future. However, predicted chance of Earth reentry in one million years is generally of the same order as the likelihood of colliding with a typical near-Earth asteroid. The only exception is the near-Hohmann transfer leg from Earth to Mars for the cargo vehicle.

- Consider Stable orbit, or disposal on interplanetary path

Mission	Disposal Starts From	Required Maneuvers	ΔV
2005 Cargo	Mars orbit, after rendezvous	Depart Mars Orbit Circularize at 1.19 A.U.	0.664 km/s 0.998
2007 Crew	Earth approach, after ECCV separates	Earth Gravity Assist Circularize at 1.19 A.U.	0 2.954
2007 Cargo	Mars orbit, after rendezvous	Depart Mars Orbit Circularize at 1.19 A.U.	0.665 1.000
2009 Crew	Earth approach, after ECCV separates	Earth Gravity Assist Circularize at 1.19 A.U.	0 3.017



2007 Crew Return

	Leg	Chance of Earth Reentry in t ≤10^6 years - % 2005/07	2007/09
Cargo	E-M	12.0	12.0
	M-D	0.0	0.0
Crew	E-M	3.8	3.8
	M-E	2.2	2.6
	E-D	1.2	1.4

E = Earth M = Mars D = Disposal

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N93-26924

ENABLER I AND II ENGINE SYSTEM DESIGN MODELING AND COMPARISONS

23 OCTOBER 1992

PRESENTED BY:

DENNIS G. PELACCIO AND CHRISTINE M. SCHEIL
SCIENCE APPLICATIONS INTERNATIONAL CORPORATION
ALBUQUERQUE, NM 87123

PRESENTED AT:

1992 NUCLEAR PROPULSION - TECHNICAL INTERCHANGE MEETING
NASA LEWIS RESEARCH CENTER
SANDUSKY, OH

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TOPICS

- OBJECTIVE/APPROACH
- ENGINE SYSTEM DESIGN/MODELING ASSUMPTIONS
- ENGINE SYSTEM SCALING/COMPARISONS
- CONCLUDING REMARKS

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OBJECTIVE/APPROACH

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OBJECTIVE / APPROACH

- **Objective:**
  - Define a Near-Term Solid-Core NTP Engine System Scaling Database
    - -- Identify/Document Unified Set of Performance, Weight and Size Scaling Data
    - -- Results Should Be Useful to Meet Initial Mission and Concept Design Study Requirements

- **Approach:**
  - Acquire/Review Past Rover/NERVA Engine Design Work
  - Assess Current Engine System Data
  - Conduct Preliminary NTP Engine System Design Trades Using the NESS Design Program
    - -- Establish Operating Range of Interest and Technology Design Approach
    - -- Design Analysis Responsibilities
      - SAIC - Engine System
      - Westinghouse - Reactor and Internal Shield (ENABLER Reactor)
  - Establish a Catalog of Enabler I and II Engine System Design for a Range Configurations and Operating Conditions

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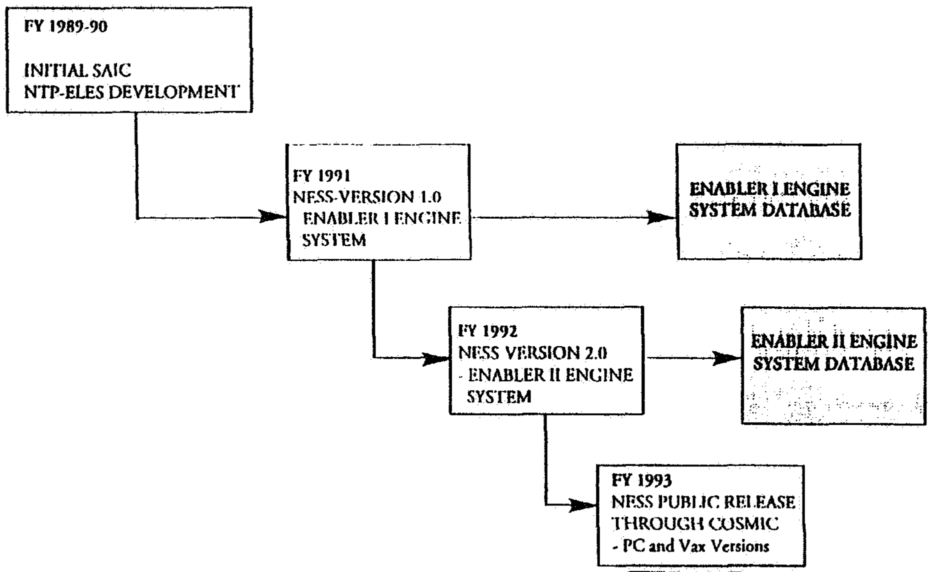
ENABLER I AND II ENGINE SYSTEM DESIGN DATABASE DEVELOPMENT CLOSELY PARALELLED NESS PROGRAM DEVELOPMENT



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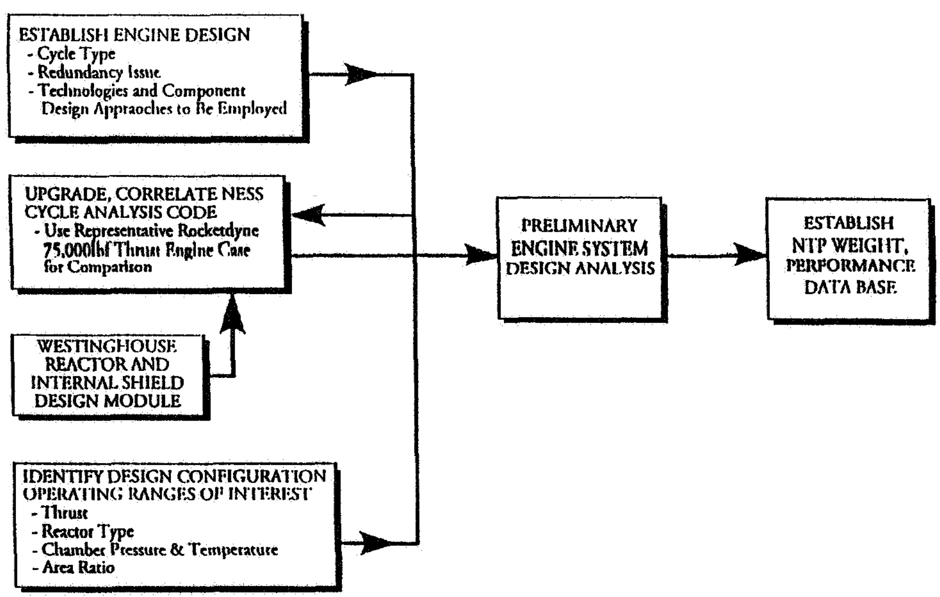
OVERALL NTP ENGINE SYSTEM ASSESSMENT APPROACH



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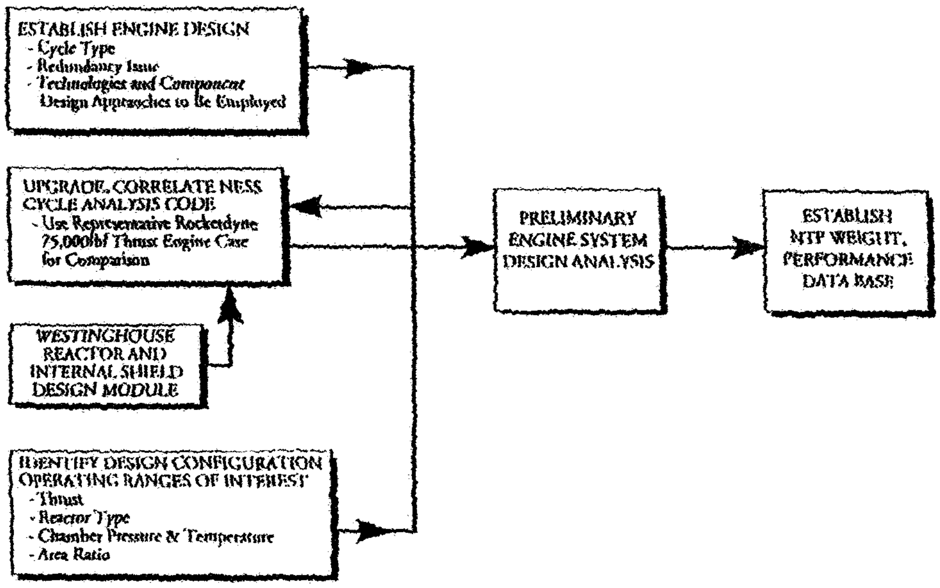
OVERALL NTP ENGINE SYSTEM ASSESSMENT APPROACH



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KEY TECHNICAL DOCUMENTS REVIEWED

- XE-Prime Engine - Final Report
- Small Nuclear Engine Final Report
- Experience Gained From the Space Nuclear Rocket Program (ROVER)
- History Summary Report
- Safe, Compact, Nuclear Propulsion - Final Report
- NERVA Preliminary Design Safety Report
- Nuclear Rocket Engine Optimization Program
- NASA Lewis Nuclear Thermal Propulsion Workshop, 1990
- "Design of Second-Generation Nuclear Thermal Rocket Engines," AIAA Report
- On-Going NASA Lewis NTP System Studies

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ENABLER REACTOR DESIGN AND OPERATING PARAMETERS EXAMINED

- Full Element/Chamber Temperature Range

  - Graphite: 2,200 - 2,500K
  - Composite: 2,500 - 2,900K
  - Carbide: 2,900 - 3,300K

- Thrust Level: 15,000-250,000 lbf

- Chamber Pressure: 500 and 1000 psia

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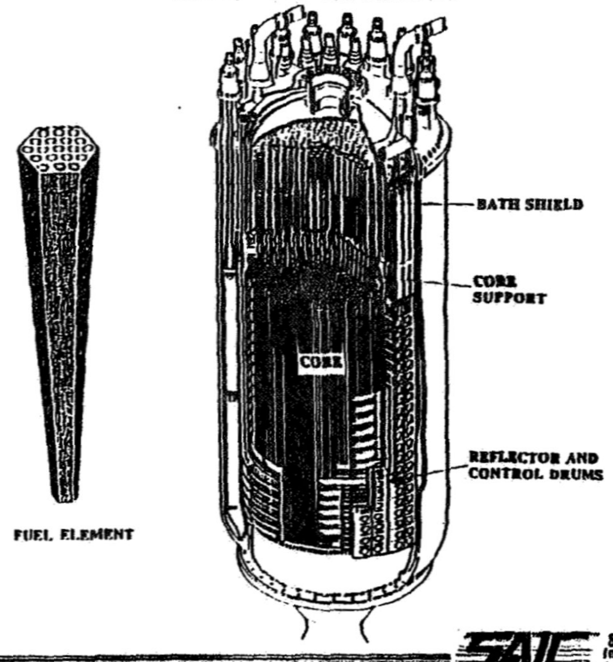
ENABLER (NERVA TYPE) NUCLEAR THERMAL ROCKET ENGINE



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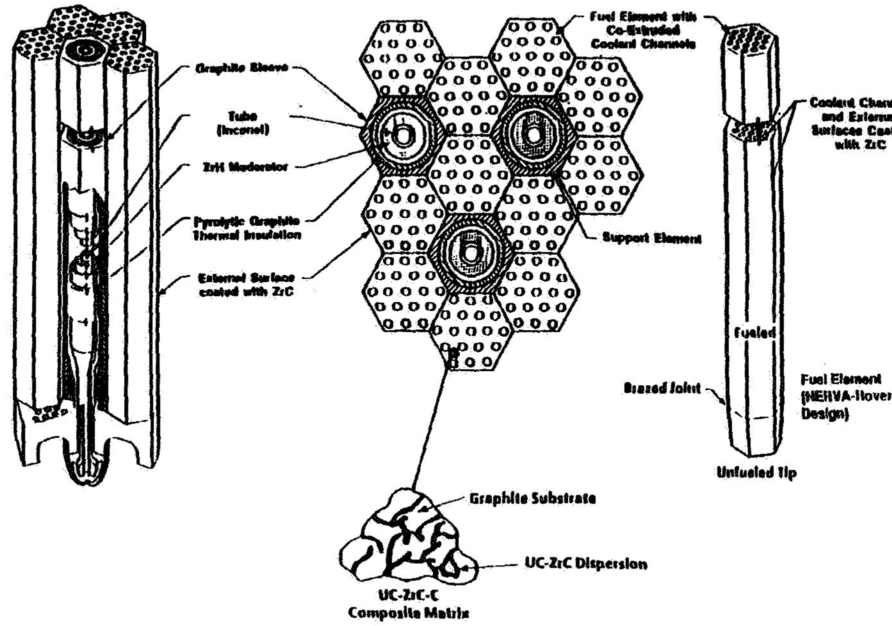
PRISMATIC FUEL ELEMENTS AND SUPPORTS



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FUEL AND SUPPORT ELEMENTS PARAMETERS

Fuel Element Composition	Graphite	Composite	Carbide
Temperature Range K	2200 - 2500	2500 - 2900	2900 - 3300
Fuel	Coated Particle	UC • ZrC Solid Solution and Carbon	(U, Zr) C Solid Solution
Coating	ZrC	ZrC	—
Unfueled Support Element Composition	Graphite	ZrC-Graphite Composite	ZrC
Unfueled Element Coating	ZrC	ZrC	—

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REACTOR PARAMETERS/CHARACTERISTICS AS A FUNCTION OF THRUST LEVEL

Thrust (1000 lbf)	15	25	>50
Reactor Power Range (MW)	275 - 400	460 - 670	920 - 6700
Fuel and Support Element Length [m (inch)]	0.89 (35)	0.89 (35)	1.32 (52)
Pressure Vessel Length [M(inch)]	2.10 (82.6)	2.13 (84)	2.58 (101.6)
Fuel Element Power (MW)	0.629	0.808	1.20
Relative Fuel Element Power Density	0.778	1.0	1.0
Ratio of Fuel Elements (N) to Support Elements	2:1	3:1	6:1
Pressure Vessel Material	Aluminum	Aluminum	Aluminum
Reflector Material	Beryllium	Beryllium	Beryllium
Internal Shield Material	BATH*/Lead	BATH/Lead	BATH/Lead

* BATH = Borated Aluminum Titanium Hydride

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RADIATION LEAKAGE LIMITS CRITERIA ASSUMED

- AT A PLANE 160 CM (63 INCHES) FORWARD OF THE CORE CENTER -

Type of Radiation	Radiation Leakage Limits Within Pressure Vessel Outside Radius
Gamma Carbon KERMA Rate	1.8×10^7 rad (c)/hr
Fast Neutron Flux	2.0×10^{12} n/cm^2 - sec, $E_n > 1.0$ MeV
Intermediate Neutron Flux	3.0×10^{12} n/cm^2 - sec, 0.4 eV $\leq E_n \leq$ 1.0 MeV
Thermal Neutron Flux	6.0×10^{11} n/cm^2 - sec, $E_n < 0.4$ ev

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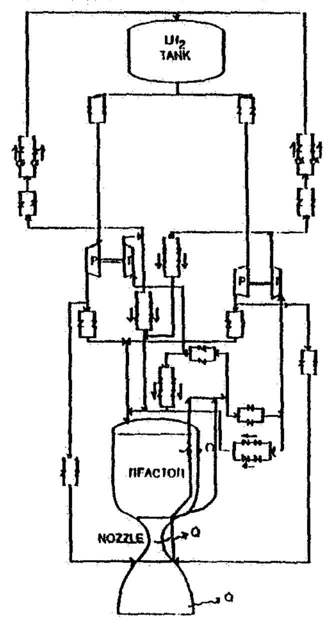
ENGINE SYSTEM ANALYSIS APPROACH/ASSUMPTIONS

- Use SAIC NESS Cycle Analysis Code
  - Check/Adjust Code to a Reasonable Test Case
    - Rocketdyne ENABLER I and ENABLER II 75,000 lbf, 1000 psia Engine System Design Selected
  - Model Engine as an Expander Cycle
  - Incorporate near-Term State-of-the-Art Technologies
  - Incorporate Dual Turbopump Feed system
    - Single Propellant turbopump With Dual Valving per Feed Leg - 80% Thrust Level Capability per Leg
    - Centrifugal Turbopumps Used
    - Boost Pumps Assumed Only for the ENABLER II Engine System
  - Use Westinghouse ENABLER Reactor System Design Model
    - Includes an Internal Shield Model
  - Examine Nozzle Ratios of 200 and 500:1



LEGEND
VALVE
CHECK VALVE
REGULATOR

EXPANDER CYCLE

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KEY NTP ENGINE DESIGN AND TECHNOLOGY FEATURES

Design/Technology Feature(s)	Comment(s)
Gimbal Engine Mount	Based on the Space Shuttle Main Engine (SSME) Design Approach
Turbopump Assembly - Dual Feed System Legs with Dual Valving – 80% Pumping Capability per Feed Leg - Centrifugal Turbopumps – Pump Material: Inconnel – Turbine Material: MAR-M246	- Based on NERVA Design Approach (Redundancy Considerations) - Based on the SSME
Solid-Core NERVA Type Reactor Design - Internal Shield	- Uses State-of-the-Art Reactor System Fuels/ Technologies/Materials (Westinghouse ENABLER Design) - Based on NERVA Design Approach and Requirements
Nozzle Assembly - 119% High Area Ratio RAO Contour – 200 and 500:1 - Three Section Assembly – Throat Region • 2:1 Upstream to 6:1 Downstream • Slotted Regen Wall Construction of Copper – Intermediate Region • 6:1 to 150:1 Downstream • Regen Cooled Inconnel Tube Bundles – Exit Nozzle Extension • 150:1 to Exit • Radiation Cooled Carbon-Carbon	- Conservative, High Performance Design, Common in the Propulsion Community - Based on the SSME and State-of-the-Art Rocket Propulsion Technology Base - Based on SSME - Based on SSME - Uses State-of-the-Art Rocket Propulsion Materials Technology Base
Miscellaneous Hardware - Propellant Lines - Valves - Electronics/Instrumentation/Processors	- State-of-the-Art Technologies Employed – Reduced Weight, Size, and Increased Capability

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EXTENSIVE NESS PROGRAM VERIFICATION CONDUCTED IN PARALELL AS THE ENABLER NTP DATABASE DEVELOPED

ENABLER I DATABASE

ENABLER II DATABASE

SAIC NTP-ELES EARLY PROTOTYPE VERSION OF NESS)

- Primarily Focused of Anchor/Verification of Non-Nuclear Components and Performance

NESS

- Primary Focused on Anchor/Verification Reactor System and Engine Integration Features

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INITIAL ENGINE COMPONENT WEIGHT COMPARISON* - 75,000 lbf NTP ENGINE CASE -

Parameter	NERVA	Rocketdyne	SAIC ELES-NTP	Adjustments/Comments
Chamber Temperature (°K)	2500	2700	2700	—
Chamber Pressures (psia)	450	1000	1000	—
Area Ratio	100	500	500	—
Specific Impulse - Vac (sec)	850	923	922.8	—
Reactor (kg)	5890	5824	5823	—
Internal Shield (kg)	1583	—	1523	—
Nozzle Assembly (kg)	1051	440	421	• ELES NTP Value Increased by 5% - Rocketdyne Weight Considered a Good Baseline
Turbopump Assembly (kg)	243	304	104	• ELES NTP Value Increased by 30% - Rocketdyne Considered Conservative for SOA Designs
Nonnuclear Support Hardware (kg) - Lines, Values, Actuators, Instrumentation Thrust Structure	2425	1815	1264	• ELES NTP Value Increased by 40% - Rocketdyne Weight Considered a Good Baseline - Scaled From Previous Design Work

* Rocketdyne uses their Mark-25 type axial turbopump (4 stages); ELES-NTP used a single-stage centrifugal pump.

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INITIAL ENGINE COMPONENT WIEGHT COMPARISON*
- 75,000 lbf NTP ENGINE CASE -

Parameter	NERVA	Rocketdyne	SAIC ELES-NTP	Adjustments/Comments
Chamber Temperature (°K)	2500	2700	2700	—
Chamber Pressures (psia)	450	1000	1000	—
Area Ratio	100	500	500	—
Specific Impulse - Vac (sec)	850	923	922.8	—
Reactor (kg)	5890	5824	5823	—
Internal Shield (kg)	1583	—	1523	—
Nozzle Assembly (kg)	1051	440	421	• ELES NTP Value Increased by 5% - Rocketdyne Weight Considered a Good Baseline
Turbopump Assembly (kg)	243	304	104	• ELES NTP Value Increased by 30% - Rocketdyne Considered Conservative for SOA Designs
Nonnuclear Support Hardware (kg) - Lines, Values, Actuators, Instrumentation Thrust Structure	2425	1815	1264	• ELES NTP Value Increased by 40% - Rocketdyne Weight Considered a Good Baseline - Scaled From Previous Design Work

* Rocketdyne uses their Mark-25 type axial turbopump (4 stages); ELES-NTP used a single-stage centrifugal pump.

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INITIAL ENGINE CYCLE PARAMETER COMPARISON*
- 75,000 lbf NTP ENGINE CASE -

Parameter	Rocketdyne	SAIC - ELES NTP
Pump Flowrate (kg/s)	36.7	36.9
Pump Discharge Pres. (psia)	1544	1538.3
Turbine Flowrate, % Pump	50	50
Turbine Inlet Temp. (°K)	555.6	555.3
Turbine Inlet Pres. (psia)	1412	1416.8
Turbine Pressure Ratio	1.25	1.295
Reactor Inlet Pres. (psia)	1130	1255.4
Reactor Power, (MW)	1645	—
Reactor Core Flowrate (kg/s)	36.7	36.9
Nozzle Chamber Temp (°K)	2700	2700
Nozzle Chamber Pres. (psia)	1000	1000
Nozzle Exit Diameter (m)	4.15	4.15
Nozzle Expansion Ratio	500	500
Specific Impulse-Vac (sec)	923	922.8
Pump Speed (rpm)	37,500	34,913

* Rocketdyne uses their Mark 25-type axial turbopump (4 stages); ELES-NTP used a single-stage centrifugal pump.

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INITIAL ENGINE CYCLE PARAMETER COMPARISON*
- 75,000 lbf NTP ENGINE CASE -

Parameter	Rocketdyne	SAIC - ELES NTP
Pump Flowrate (kg/s)	36.7	36.9
Pump Discharge Pres. (psia)	1544	1538.3
Turbine Flowrate, % Pump	50	50
Turbine Inlet Temp. (°K)	555.6	555.3
Turbine Inlet Pres. (psia)	1412	1416.8
Turbine Pressure Ratio	1.25	1.295
Reactor Inlet Pres. (psia)	1130	1255.4
Reactor Power, (MW)	1645	—
Reactor Core Flowrate (kg/s)	36.7	36.9
Nozzle Chamber Temp (°K)	2700	2700
Nozzle Chamber Pres. (psia)	1000	1000
Nozzle Exit Diameter (m)	4.15	4.15
Nozzle Expansion Ratio	500	500
Specific Impulse-Vac (sec)	923	922.8
Pump Speed (rpm)	37,500	34,913

* Rocketdyne uses their Mark 25-type axial turbopump (4 stages); ELES-NTP used a single-stage centrifugal pump.

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CYCLE PARAMETER COMPARISON*
- 75,000 lbf ENABLER I, EXPANDER CYCLE -

Parameter	Rocketdyne	SAIC - ELES NTP	SAIC NESS
Total Flowrate (kg/s)	36.7	36.9	37.27
Pump Discharge Pres. (psia)	1,544	1,538.3	2,298.3
Turbine Flowrate, % Pump	50	50	50
Turbine Inlet Temp. (°K)	555.6	555.3	622.3
Turbine Inlet Pres. (psia)	1,412	1,416.8	1,969.0
Turbine Pressure Ratio	1.25	1.295	1.739
Reactor Inlet Pres. (psia)	1,130	1,255.4	1,132.1
Reactor Power, (MW)	1,645	-	1,587
Reactor Core Flowrate (kg/s)	36.7	36.9	36.2
Nozzle Chamber Temp (°K)	2,700	2,700	2,700
Nozzle Chamber Pres. (psia)	1,000	1,000	1,000
Nozzle Exit Diameter (m)	4.15	4.15	4.22
Nozzle Expansion Ratio	500	500	500
Specific Impulse-Vac (sec)	923	922.8	912.9
Pump Speed (rpm)	37,500	34,913	40,583

* Rocketdyne uses their Mark 25 type axial turbopump (4 stages); SAIC ELES-NTP used a single-stage centrifugal pump; SAIC NESS, Sample Case No. 8, uses a 5-stage axial pump.

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CYCLE PARAMETER COMPARISON*
- 75,000 lbf ENABLER I, EXPANDER CYCLE -

Parameter	Rocketdyne	SAIC ELES-NTP	SAIC NESS
Specific Impulse - Vac (sec)	923	922.8	912.9
Reactor (kg)	5,824	5,823	4,783
Internal Shield (kg)	—	1,523	1,108
Nozzle Assembly (kg)	440	421	535
Turbopump Assembly (kg)	304	104	221
Nonnuclear Support Hardware (kg) - Lines, Values, Actuators, Instrumentation Thrust Structure	1,815	1,264	1,493

* Rocketdyne uses their Mark 25 type axial turbopump (4 stages); SAIC ELES-NTP used a single-stage centrifugal pump; SAIC NESS, Sample Case No. 8, uses a 5-stage axial pump.

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CYCLE PARAMETER COMPARISON*
- 75,000 lbf ENABLER I, EXPANDER CYCLE -

Wall Temperature (°R)	Barrier Temperature (°R)	Isp (Sec.)	Fuel Film Cooling Fraction
1460	1630	912.9	0.03
1800	2106	915.9	0.03
2000	2429	917.5	0.02
2400	2892	919.4	0.02
2800	3418	921.2	0.02
3000	3651	921.9	0.02
3200	3864	922.4	0.02

* Core Temperature = 4000°R (2700°K)

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DESIGN CASE COMPARISION OBSERVATIONS

- NESS Design Exhibits 1% Lower Peformance Than Other Designs
  - NESS Model More Accurately Predicts Nozzle Cooling Losses-Upstream Film Cooling Required to Meet Maximum Wall Temperature Requirements
- Integrated Reactor/Engine System Design Effects Accounted for in the NESS Design
  - Sized to Take Into Account Heat Captured by the Coolant Before It Enters the Reactor
  - Corresponds to Some Difference in Cycle Pressures, Temperatures, and Turbopump Operating Parameters
- Other Weight Differences From Improvements in NESS Weight Correlations
  - 3-Section Nozzle Design
  - Non-Nuclear Auxiliary Components
  - Update H_2 Properties

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ENGINE SYSTEM SCALING/COMPARISONS

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EXTENSIVE ENABLER I AND II ENGINE SYSTEM DESIGN DATABASE HAS BEEN DEVELOPED

- Database Covers A Large Engine Design/ Operating Paramenters
  - Fuel Type/Chamber Temperature
  - Thrust level
  - Chamber Pressure
  - Nozzle Area Ratio
- Top-level Design Scaling Trends Produced for the ENABLER I Engine System
  - Little Design Trend Analysis Conducted to Date on the Eanbler II Database
- All Engine Summary Design Data is Cataloged and is Available Through NASA Lewis

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ENABLER ENGINE SYSTEM DESIGN TRADE SPACE ANALYZED

Fuel Type/Chamber Temperature (oK)	Thrust (lbf)	Chamber Pressure (psia)	Nozzle Area Ratio
Graphite/2500 Composite/2700 Carbide/3100	15,000 40,000 75,000 125,000 200,000 250,000	500 750 1,000 1,500 2,000	200:1 500:1

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ENABLER I DESIGN SCALING TRENDS

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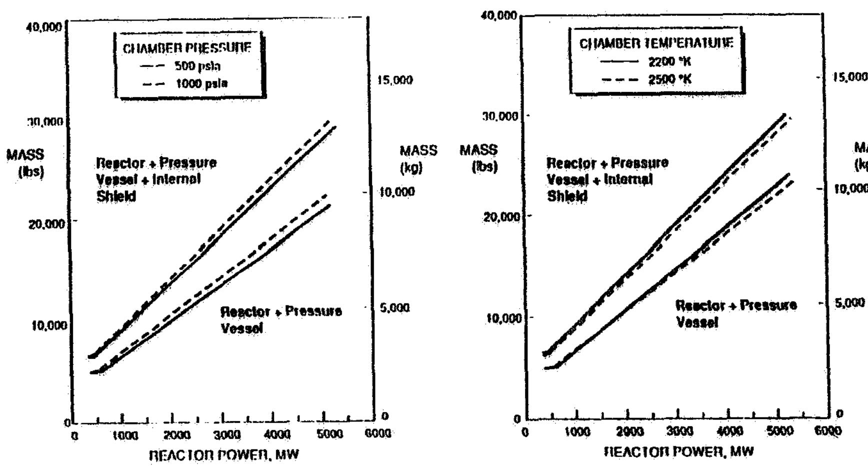
REACTOR AND INTERNAL SHIELD MASS SCALING
- Graphite Fuel -



CHAMBER TEMPERATURE = 2500 °K

CHAMBER PRESSURE = 1000 psia

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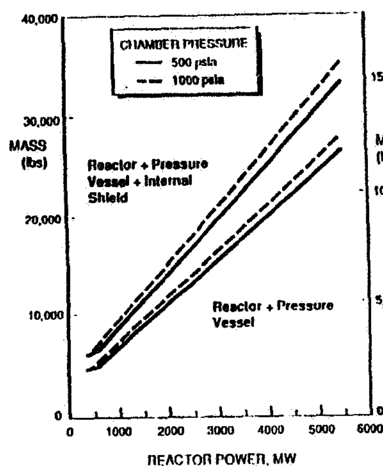
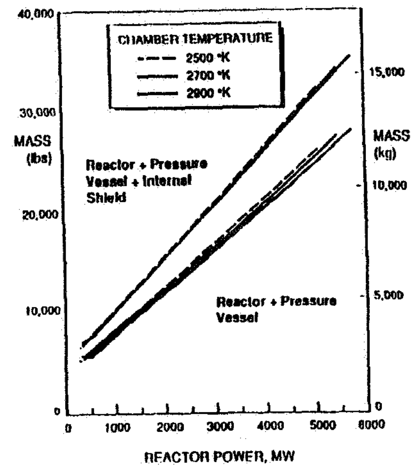
REACTOR AND INTERNAL SHIELD MASS SCALING
- Composite Fuel -



CHAMBER TEMPERATURE = 2700 °K



CHAMBER PRESSURE = 1000 psia

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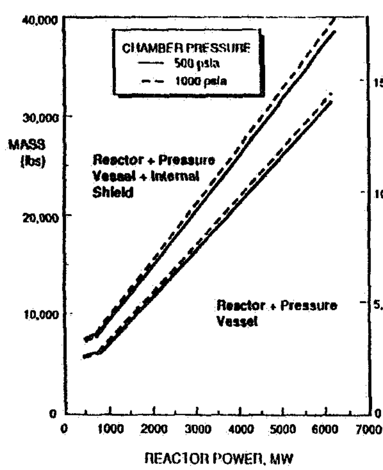
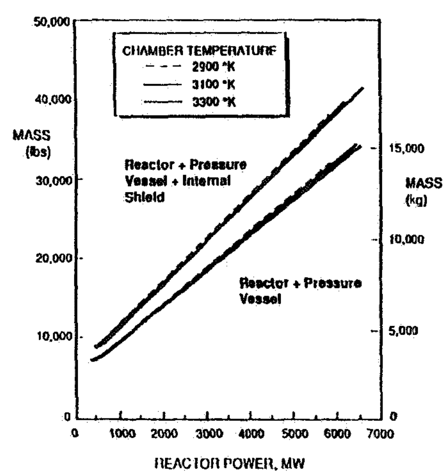
REACTOR AND INTERNAL SHIELD MASS SCALING
- Carbide Fuel -



CHAMBER TEMPERATURE = 3100 °K



CHAMBER PRESSURE = 1000 psia

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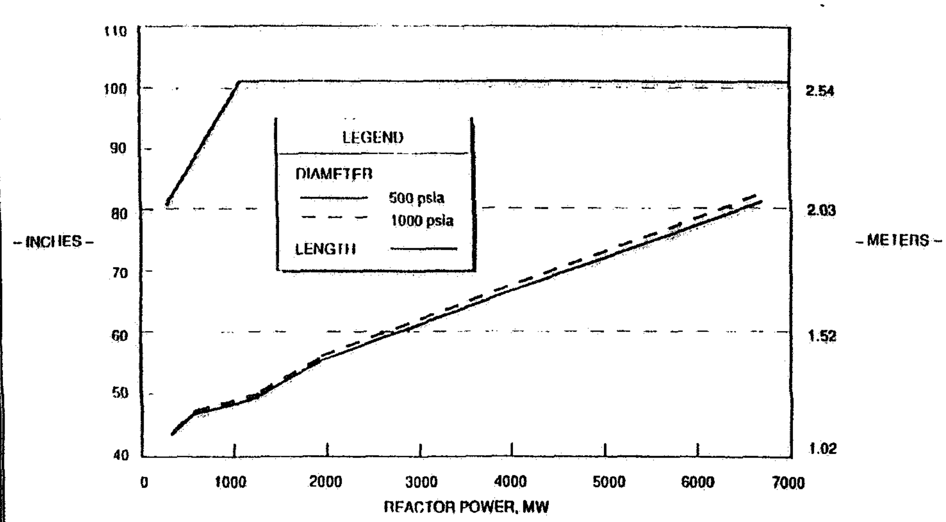
REACTOR PRESSURE VESSEL DIMENSIONS AS A FUNCTION OF POWER AND CHAMBER PRESSURE



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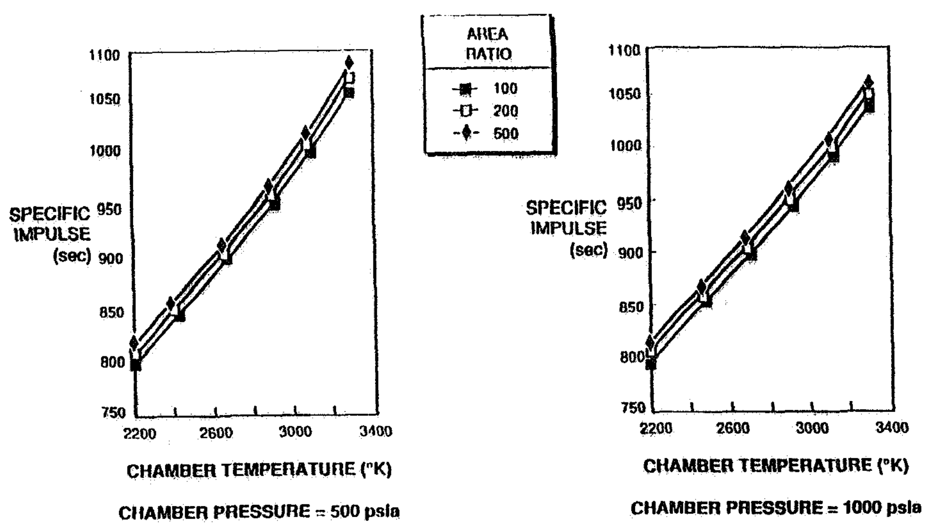
TYPICAL NTP ENGINE PERFORMANCE AS A FUNCTION OF CHAMBER PRESSURE, TEMPERATURE AND AREA RATIO - 75,000 lbf Thrust -



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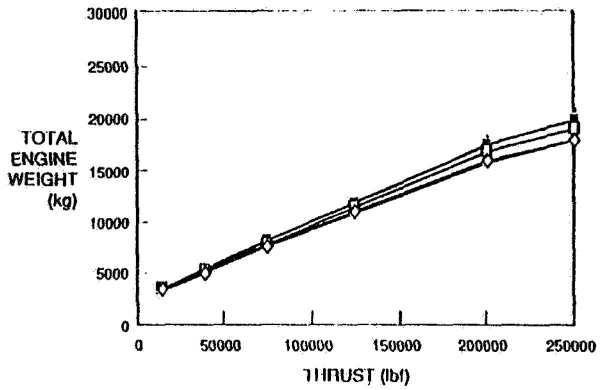
NTP ENGINE WEIGHT AS A FUNCTION OF THRUST, CHAMBER PRESSURE, AND AREA RATIO
- Graphite Fuel, Chamber Temperature - 2500 K -



KEY		
SYMBOL	CHAMBER PRESSURE (psia)	AREA RATIO
-■-	500	500
-□-	1000	500
-◆-	500	200
-◇-	1000	200

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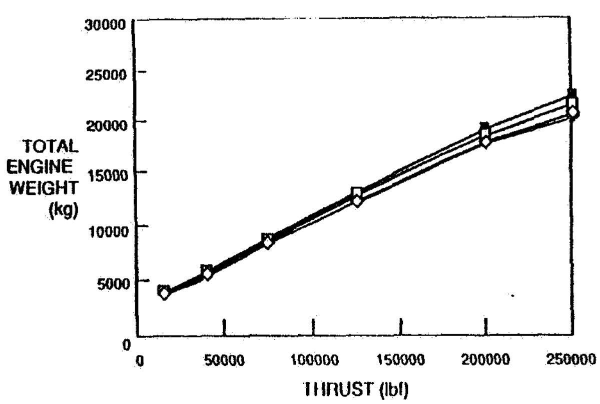
NTP ENGINE WEIGHT AS A FUNCTION OF THRUST, CHAMBER PRESSURE, AND AREA RATIO
- Composite Fuel, Chamber Temperature - 2700 K -



KEY		
SYMBOL	CHAMBER PRESSURE (psia)	AREA RATIO
-■-	500	500
-□-	1000	500
-◆-	500	200
-◇-	1000	200

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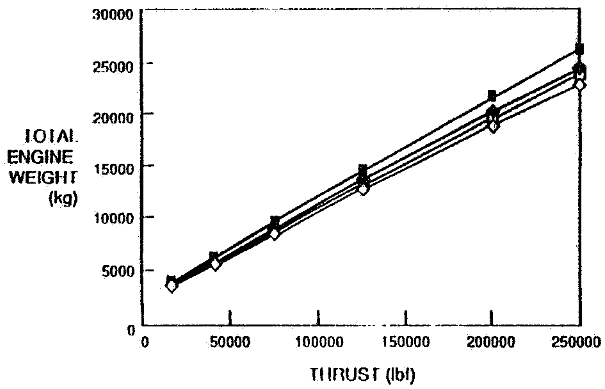
NTP ENGINE WEIGHT AS A FUNCTION OF THRUST, CHAMBER PRESSURE, AND AREA RATIO

- Carbide Fuel, Chamber Temperature - 3100 K -



KEY		
SYMBOL	CHAMBER PRESSURE (psia)	AREA RATIO
-■-	500	500
-□-	1000	500
-◆-	500	200
-◇-	1000	200

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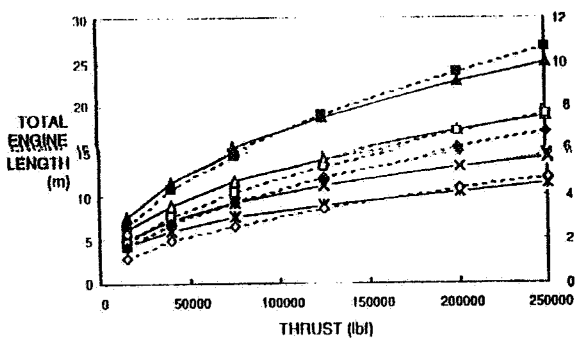
NTP ENGINE SYSTEM AS A FUNCTION OF THRUST,

- Graphite and Composite Fuel Engines -



KEY			
DIMENSION	SYMBOL	CHAMBER PRESSURE (psia)	AREA RATIO
Length	★	500	500
	☆	1,000	
	✕	500	200
	✱	1,000	
Diameter	■	500	500
	□	1,000	
	◆	500	200
	◇	1,000	

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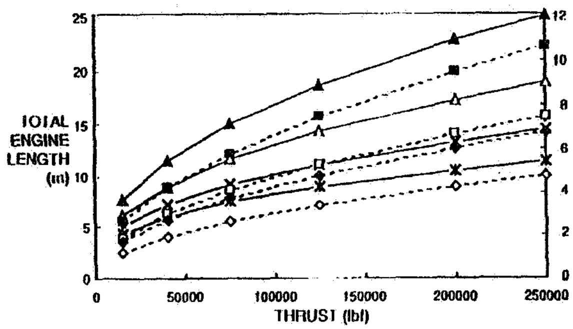
NTP ENGINE SYSTEM AS A FUNCTION OF THRUST,
- Carbide Fuel Engines -



KEY			
DIMENSION	SYMBOL	CHAMBER PRESSURE (psia)	AREA RATIO
Length	▲	500	500
	△	1,000	
	✕	500	200
	✕	1,000	
Diameter	■	500	500
	□	1,000	
	◆	500	200
	◇	1,000	

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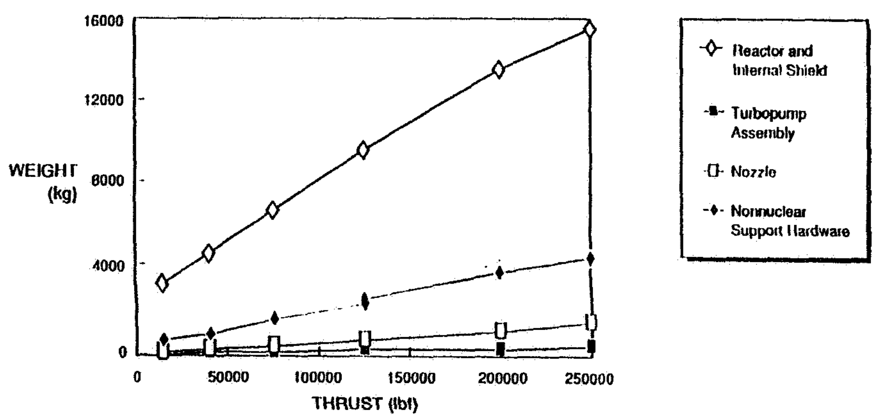
NTP ENGINE SUBSYSTEM WEIGHT BREAKDOWN AS A FUNCTION OF THRUST,
- Composite Fuel, T_c = 2700 K, P_c - 1000psia, ε = 500:1 -



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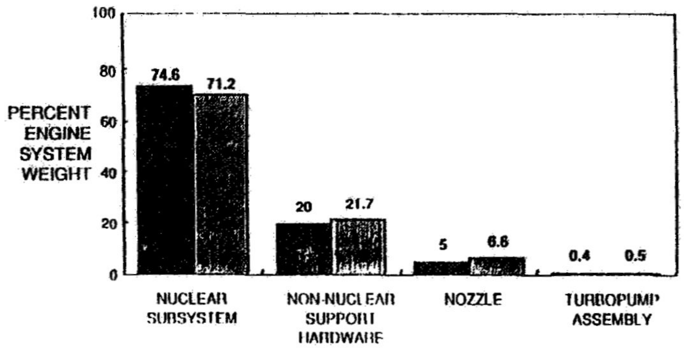
NTP ENGINE SUBSYSTEM PERCENT WEIGHT DISTRIBUTION FOR TWO THRUST LEVELS

- Composite Fuel, T_c = 2700 K, P_c = 1000 psia, ε = 500:1



KEY		
SYMBOL	ENGINE THRUST (lbf)	ENGINE WEIGHT (kg)
■	75,000	8,816
▨	250,000	22,410

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BASELINE NTP ENGINE DESCRIPTION*

- 75,000 LBF THRUST, EXPANDER CYCLE, COMPOSITE FUEL, T_c = 2700 K, P_c = 1000psia, ε = 500:1 -

Component	Features		Number
Reactor	• Reactor + Internal Shield Weight • Fuel Type - Composite • Case Material - Aluminum • Reactor Diameter • Reactor Length • Fuel Mass Flow Rate • Reactor Exit/Nozzle Entrance - Exit Chamber Pressure - Temperature	6576 kg (14,500 lbm)* √ 1.32 m (52 in) 2.59 m (102 in) 36.9 kg/s (81.3 lbm/s) 6895 kPa (1000 psia) 2,700 K (4,860°R)	1
Nozzle	• Nozzle Weight • Nozzle Material - Slotted Regen Wall Construction of Copper to Area Ratio of 6:1 - Inconel Tube Bundles to Area Ratio of 150:1 • Extension Material - Carbon • Regeneratively Cooled by Propellant to an Area Ratio of 150:1 • Nozzle Length • Throat Diameter • Exit Diameter • Area Ratio • Delivered Vacuum Isp • Delivered Thrust	442 kg (975 lbm) √ √ √ 823 cm (324 in) 18.8 cm (7.4 in) 415.8 cm (163.7 in) 500:1 9043 N x sec/kg (923 sec) 333.6 kN (75,000 lbf)	1
Main Pump Turbine	• Turbine Weight • Material MAR - M246 • No. of Full Admission Turbine Stages • Efficiency • Pressure Ratio • Diameter • Turbine Speed	15.9 kg (35 lbm) √ 2 0.70 1.995 16.0 cm (6.8 in) 39,028 rpm	2

* Total Component Weight - Typical

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BASELINE NTP ENGINE DESCRIPTION*

- 75,000 LBF THRUST, EXPANDER CYCLE, COMPOSITE FUEL, T_c = 2700 K, P_c = 1000psia, ε = 500:1 - (CONT.)

Component	Features		Number
Main Fuel Pump	• Main Pump Weight • Material - Inconnel • Single-Stage Centrifugal Pump • Pressure Rise • Pump Speed • Pump Diameter • Pump Efficiency • Pump Horsepower	44.6 kg (98.4 lbm) √ √ 10,260 kPa (1488 psia) 39,028 rpm 25.7 cm (10.1 in) 0.715 8143 HP	2
Misc. Hardware Weights	• Thrust Mount • Thrust Support Hardware • Engine Lines • Main Valve • TPA Ignition • Gimbal System	737 kg (1624 lbm) 573 kg (1263 lbm) 91.9 kg (202.7 lbm) 182.6 kg (402.6 lbm) 15.7 kg (34.7 lbm) 34.9 kg (76.9 lbm)	1 1 2 4 2 1
Subtotal	• Total Nonnuclear Weight (TPA + Misc. Hardware + Nozzle) • Margin (2%)	2196 kg (4843 lbm) 44 kg (96.9 lbm)	
Total Engine System		8816 kg (19,440 lbm)	

* Total Component Weight - Typical

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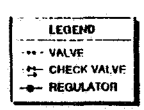
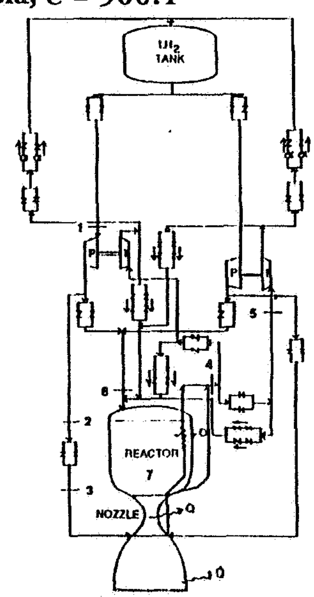
BASELINE NTP ENGINE CYCLE, OPERATING PARAMETERS

- 75,000 lbf Thrust, Composite Fuel, T_c = 2700 K, P_c = 1000psia, ε = 500:1 -

STATION	PRESSURE (psia)	TEMP (°K)	$\dot{m}$ (kg/s)
1	50	22.2	18.4
2	1538	40.3	18.4
3	1467	40.3	18.4
4	1417	555.3	38.9
5	1404	555.3	18.4
6	1255	533.6	36.9
7	1000	2700.0	36.9





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BASELINE NTP ENGINE CYCLE, OPERATING PARAMETERS

- 75,000 lbf Thrust, Expander Cycle, Composite Fuel, T_c = 2700 K, P_c = 1000psia, ε = 500:1 -

ENGINE SUMMARY

EXPANDER CYCLE				
ENABLER II				
CENTRIFUGAL PUMPS				
THRUST LEVEL =	75000.0	lbf	333600.0	N
CHAMBER PRESSURE =	1000.0	psia	6895.0	kPa
CHAMBER TEMPERATURE =	4860.0	deg R	2700.0	deg K
NOZZLE EXIT AREA RATIO =	500.0		500.0	
NUMBER OF FEED LEGS =	2		2	
TOTAL PROPELLANT FLOWRATE =	83.1	lbm/s	37.7	kg/s
REACTOR				
COMPOSITE FUEL				
FUEL SCALING FACTOR	0.07		0.07	
REACTOR WEIGHT	6151.1	lbm	2790.0	kg
SHIELD WEIGHT	2424.3	lbm	1099.5	kg
PRESSURE VESSEL DIA.	45.2	in	114.7	cm
PRESSURE VESSEL LENGTH	87.0	in	221.0	cm
CORE PROPELLANT MASS FLOW	80.7	lbm/sec	36.6	kg/sec
NOZZLE				
CONVERGING NOZZLE WEIGHT	167.5	lbm	76.0	kg
NOZZLE EXTENSION WEIGHT	59.5	lbm	27.0	kg
SECOND NOZZLE EXTENSION WEIGHT	228.1	lbm	103.5	kg
TOTAL NOZZLE WEIGHT	455.2	lbm	206.4	kg
AREA RATIO	500.0		500.0	
THROAT DIAMETER	7.5	in	19.0	cm
EXIT DIAMETER	105.6	in	268.3	cm
NOZZLE LENGTH	163.0	in	414.1	cm
DELIVERED VACUUM ISP	902.5	sec	8844.4	N-sec/kg
DELIVERED THRUST	75000.0	lbf	333600.0	N
TURBOPUMP ASSEMBLY (TOTAL FOR ALL FEED LEGS)				
MAIN PROP. TURBOPUMP WT	400.0	lbm	181.5	kg
PROPELLANT BOOST PUMP WT	64.4	lbm	29.2	kg
MAIN OX PUMP WEIGHT	0.0	lbm	0.0	kg
TPA IGNITION WEIGHT	32.2	lbm	14.6	kg
BLEED LINE/VALVE WEIGHT	0.0	lbm	0.0	kg
MISC. HARDWARE WEIGHTS				
THRUST MOUNT	1676.0	lbm	760.5	kg
SUPPORT HARDWARE	610.1	lbm	276.4	kg
ENGINE LINES	215.0	lbm	97.6	kg
MAIN VALVE	382.0	lbm	173.0	kg
GIMBAL + POWER SUPPLY	302.8	lbm	137.3	kg
MARGIN (2.0%)	83.1	lbm	37.7	kg
TOTAL NONNUCLEAR WEIGHT	4236.1	lbm	1921.1	kg
TOTAL ENGINE SYSTEM				
TOTAL ENGINE WEIGHT	12811.0	lbm	5810.2	kg
TOTAL ENGINE WEIGHT WITHOUT SHIELD	10387.2	lbm	4710.0	kg
THRUST/WEIGHT RATIO WITH SHIELD	5.9	lbf/lbm	57.4	N/kg
THRUST/WEIGHT RATIO WITHOUT SHIELD	7.2	lbf/lbm	70.8	N/kg
REACTOR SAFETY ROD WT.-LAUNCH ONLY	310.9	lbm	141.0	kg
TOTAL ENGINE LAUNCH WEIGHT	13122.4	lbm	5951.2	kg
TOTAL ENGINE LAUNCH WT. W/O SHIELD	10698.1	lbm	4851.8	kg
PUMP-OUT CONDITIONS				
PUMP-OUT THRUST	60000.0	lbf	266880.0	N
PUMP-OUT CHAMBER PRESSURE	800.0	psia	5516.0	kPa
PUMP-OUT ISP	903.1	sec	8856.0	N-sec/kg
PUMP-OUT CHAMBER TEMPERATURE	4860.0	deg R	2700.0	deg K
OVERALL DIMENSIONS				
OVERALL ENGINE LENGTH =	349.0	in	886.7	cm
OVERALL ENGINE DIAMETER =	105.6	in	268.3	cm

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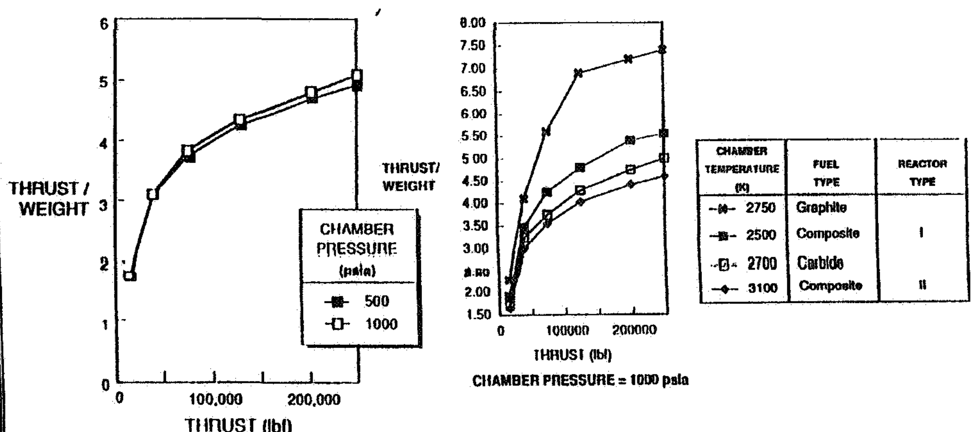
NTP ENGINE THRUST-TO-WEIGHT RATIO AS A FUNCTION OF THRUST

- ε = 500:1 -



COMPOSITE FUEL
CHAMBER TEMPERATURE = 2700° K

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CONCLUDING REMARKS

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CONCLUDING REMARKS

- A Near-Term (ENABLER I and II) NTP Solid-Core System Database has Been Established
  - Based on the Well Documented/Anchored SAIC NESS Design Program
  - Incorporates Westinghouse's SOA Reactor System Design Correlations
  - Database is Organized, Documented and is Available Through NASA Lewis

- Future Recommendations
  - Perform a Comparative Assessment of the Database
    - -- Past Engineering Data Generated
    - -- Technology Sensitivity Studies
  - Initiate a Similar Study Activity With Engine Systems Using Different Reactor Types

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N93-26925

GENERAL DYNAMICS
Space Systems Division

Clustered Engine Study Team

Kyle Shepard	Study Manager
Paul Sager	Propulsion
Sid Kusunoki	Vehicle Design
John Porter	Systems Analysis
Al Campion	Mass Properties
Gunner Mouritzan	Propulsion
Will Glunt	Trajectory/Performance
George Vegter	Guidance and Control
Rob Koontz	Guidance and Control

GENERAL DYNAMICS
Space Systems Division

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Run Tank Based Systems....................12
Boost Pump Based Systems....................21
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Summary and Conclusions....................43

CLUSTERED NTR STUDY INTRODUCTION

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

OVERVIEW

- Study Introduction
- Mission Description
- Reference Vehicle Design
- Propulsion System
- Systems Analysis
- Summary And Conclusions

KMS 1/7/92

The presentation will cover several topics which together encompass this preliminary assessment of nuclear thermal rocket engine clustering. The study objectives, schedule, flow and groundrules are covered. This is followed by the NASA groundruled mission and our interpretation of the associated operational scenario. The NASA reference vehicle is illustrated, then we zoom in on the four propulsion system options examined in this study. Each propulsion system's preliminary design, fluid systems, operating characteristics, thrust structure, dimensions and mass properties are detailed as well as the associated key propulsion system/vehicle interfaces. A brief series of systems analysis will also be covered including: thrust vector control requirements, engine out possibilities, propulsion system failure modes, surviving system requirements and technology requirements. The presentation concludes with an assessment of vehicle/propulsion system impacts due to the lessons learned in this study.

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CLUSTERED NTR STUDY OBJECTIVE

"To develop a top level assessment of the feasibility of clustering Nuclear Thermal Rocket engines."

A NASA reference vehicle and mission scenario were given.

The approach then was to develop four propulsion system designs that could be used as reference configurations for future engineering assessments.

The Study addresses:

- **Two and three engine propulsion system designs with either boost pumps or run tanks for engine start up**
- **Thrust Vector Control (TVC) Requirements**
- **Engine out possibilities**
- **Propulsion system Failure modes**
- **Technology requirements**

KMS 1/7/92

The objective of the study was to develop propulsion system designs that could be integrated with the provided reference vehicle and fly the provided reference mission. Four propulsion system options were developed using two and three engines with either boost pumps or run tanks for engine start up. Our intent was to develop propulsion systems with a cluster of NTR engines that could be used as reference configurations for future systems optimization. In doing this we considered the following system issues: TVC requirements, Engine out possibilities, propulsion system failure modes and technology development requirements.

CLUSTERED NTR STUDY

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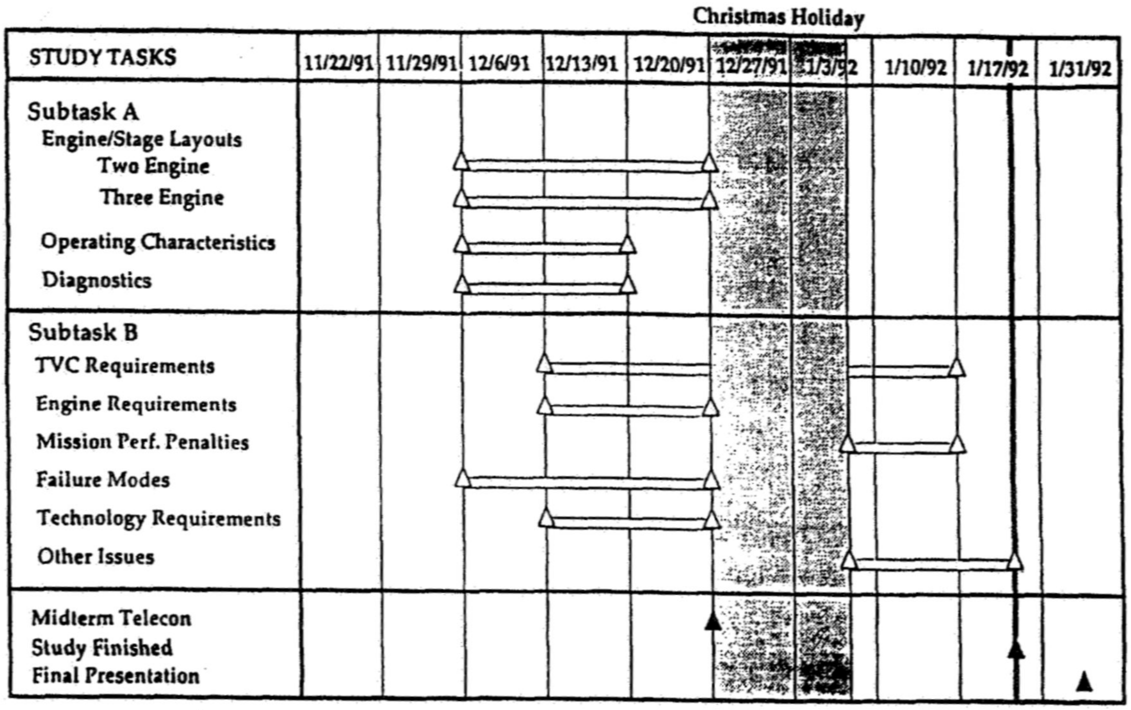
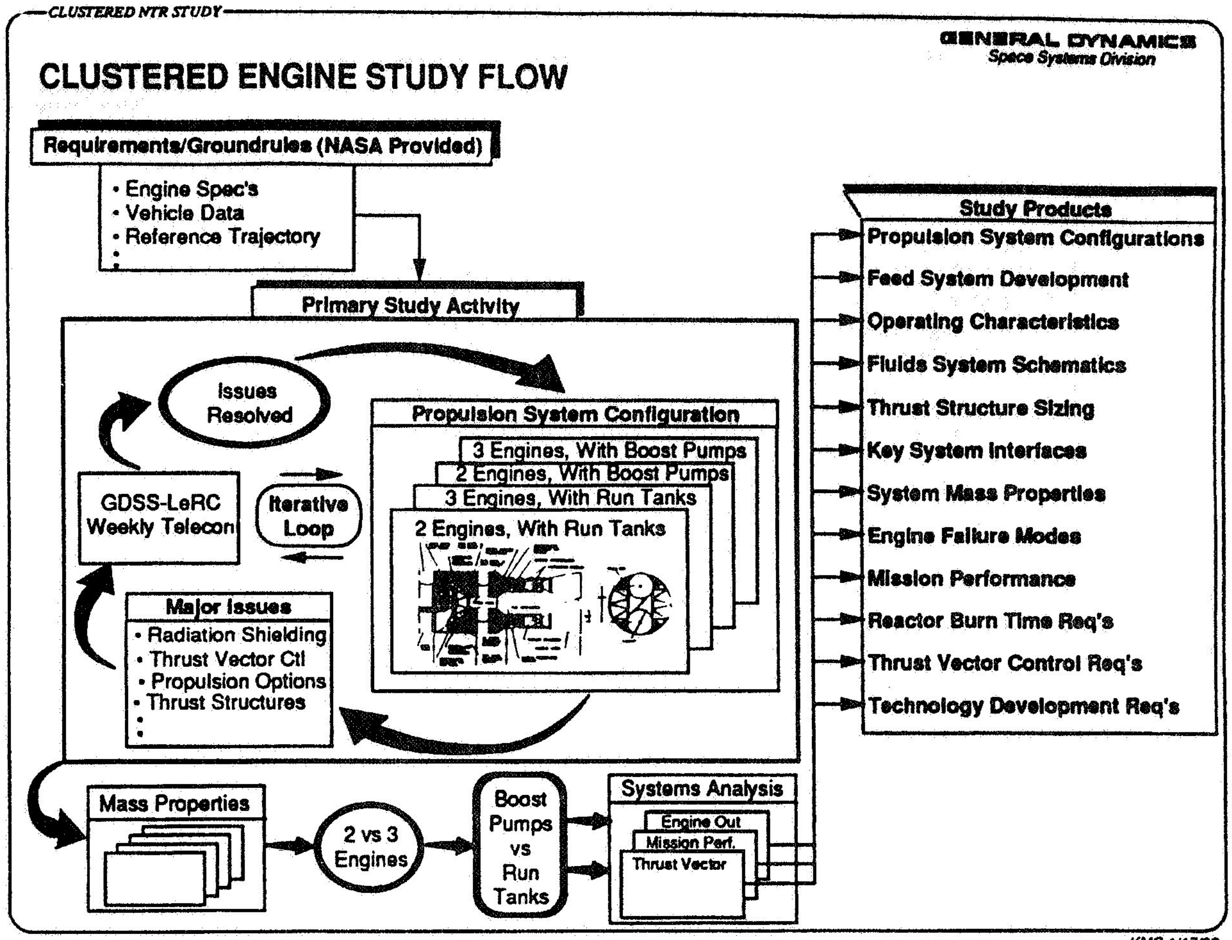
STUDY SCHEDULE



KMS 1/7/92

the study was a five week effort beginning the first week of December 1991, with a christmas holiday in the middle and ending on Jan. 15, 1992. The propulsion system preliminary designs and systems analysis were primarily completed in the first three weeks of the study. The remainder was used for analysis and design iterations as well as presentation preparation.



The study was initiated with a series of NASA LeRC provided groundrules and requirements. These were provided in appendix form and served as the point of departure for the NTP vehicle, mission, engine and propulsion system.

The primary study activity consisted of developing preliminary propulsion system designs for two and three engine propulsion systems with either run tanks or boost pumps for engine start up. As these propulsion systems were developed, several design issues arose. Design issues were addressed at LeRC-GDSS weekly telecons where issues were raised, resolved and the resulting decision(s) were applied to the design work. This iteration process continued throughout the study.

Upon completion of the design phase, mass properties were developed and a series of systems analysis took place. The systems work concentrated on issues relating to the engine out scenario. This analysis allowed us to quantify thrust vector control, reactor burn time and technology requirements as well as assess impacts to the vehicle such as mission performance penalties and failure modes.

MISSION DESCRIPTION

CLUSTERED NTR STUDY

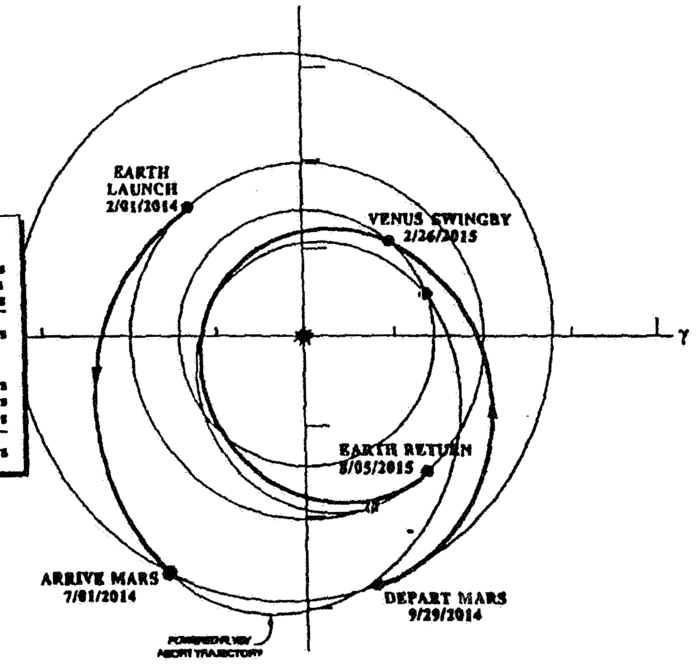
GENERAL DYNAMICS
Space Systems Division

REFERENCE MARS TRAJECTORY

SHORT-DURATION PILOTED MSSION
(2014 Opportunity with Venus Swingby)

MISSION TIMES

Nominal	
OUTBOUND	150 days
STAY	90 days
RETURN	310 days
TOTAL MISSION	550 days
Abort	
OUTBOUND	150 days
STAY	0 days
RETURN	378 days
TOTAL MISSION	528 days



KMS 1/17/92

The reference trajectory is a short opposition type trajectory. It is developed around the 2014 mission opportunity and includes a Venus swingby on Earth return. The outbound leg lasts 150 days and includes three perigee burns for Earth departure. Upon arrival at Mars, the crew performs a surface science mission lasting 90 days. The Earth return leg lasts 310 days and includes a single burn for Mars departure. Note that there is a robust Mars powered flyby abort mode available should a problem occur after Trans Mars Injection(TMI) or before Mars orbit capture(MOC).

TMI, MOC and Trans Earth Injection (TEI) burns were considered in our engine out/mission performance analysis. We consider cases for either 1 or 2 engines out for the boost pump and run tank based propulsion system options.

REFERENCE VEHICLE DESIGN

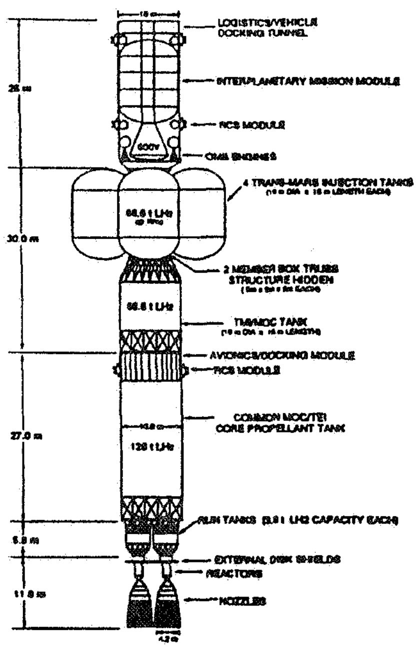
CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

REFERENCE MARS TRANSFER SYSTEM

- Vehicle assessed from core tank aft
- Primary Vehicle Modifications:
  - Core Propellant Tank (For Boost Pump Config's Only)
  - Thrust Structures
  - Run Tanks
  - Reactor Shields
  - Reactors
  - Nozzle Extension

2014 NTR-Powered Piloted Mars Vehicle (Modular Design)
"No MEV" Split/Sprint Mission Mode
(2 - 75 klbf NERVA-Derivative Engine/Composite Fuel/Isp=925s)

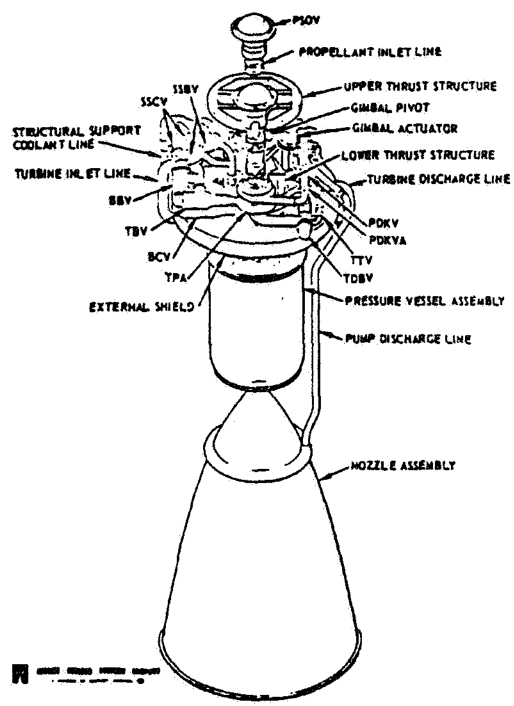


KMS 1/17/92

Our analysis concentrates on the vehicle elements from the core tank and aft. The core propellant tank is resized for the boost pump case. Sizing is based on a combination of a fully integrated propulsion system launch requirement on either STS or Titan IV. This scenario enables a more traditional intertank adapters/thrust structure. In moving from the two to three engine case, the run tanks are re-sized to take advantage of a reduced requirement for propellant volume at start. The reactor shields are modified to remove the center shield section and include a side shields. This is done to reduce shielding mass. The reactors themselves are also reduced in size due to the reduced thrust requirement on the three engine case. Lastly, an engine without a nozzle extension was groundruled.

GENERAL DYNAMICS
Space Systems Division

REFERENCE 75 klbf NERVA ENGINE



KMS 1/17/92

The reference engine is a NERVA "full flow" concept developed by Altseimer of Aerojet Nuclear Systems Company, circa 1971. It is an engine typical of that era. For the two engine cases, the reference 75 klbf engine was used, the three engine cases utilized a scaled down version of this engine sized at 50 klbf.

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM PRELIMINARY DESIGN

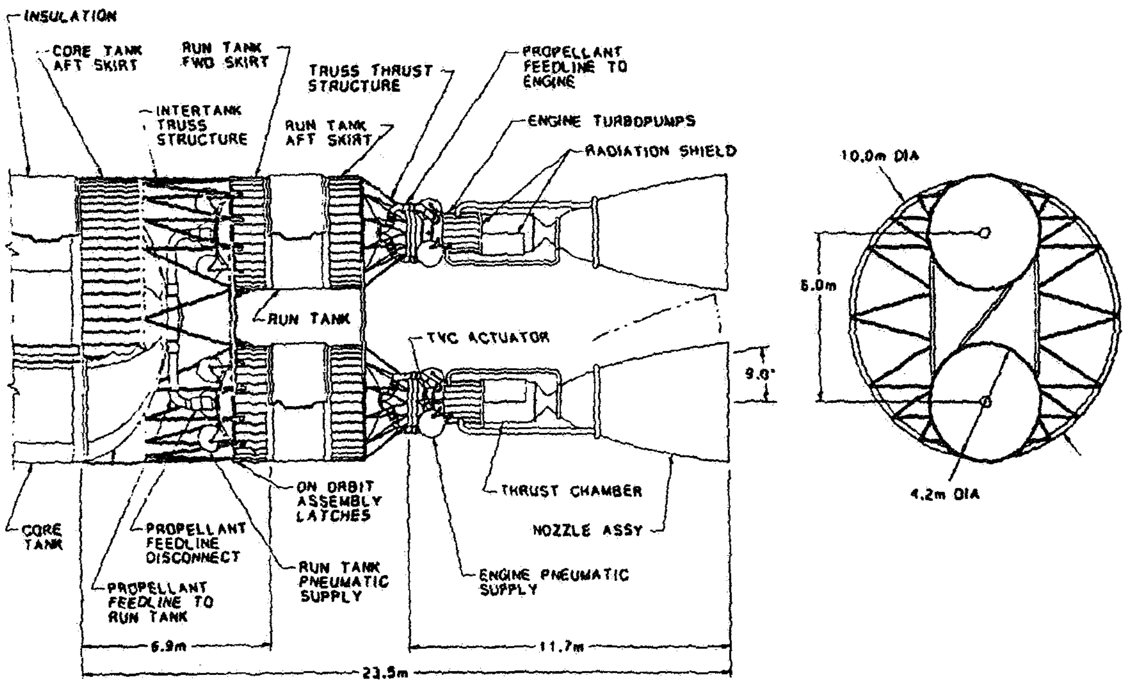
CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

RUN TANK BASED SYSTEMS

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM R-2 DESCRIPTION & DIMENSIONS



SHK 1/9/92

This configuration utilizes two 75 klb thrust NERVA nuclear thermal rocket engines with separate run tanks. The run tanks are used to minimize pressurization gas requirements for engine start. Gaseous helium for pressurizing the run tanks is supplied by high pressure bottles located above the run tanks. Once engine start is achieved, hydrogen gas is bled from the engines and used to pressurize the core tank. After the core tank is sufficiently pressurized, propellant from the core tank is fed through the run tanks to continue to feed the engines. At the end of each burn, the run tanks may be filled to capacity to repeat the procedure for the next engine start.

The run tank, engine, and thrust structure combine to form the propulsion module. The propulsion module is launched seperately from the rest of the vehicle and is coupled to the core tank on orbit. Fluid system and electrical disconnects and structural latches are provided to allow for on orbit coupling of the propulsion module to the core tank.

An aluminum-lithium (Al-Li) tubular intertank truss structure transfers the thrust from the propulsion modules to the core tank. Lateral Al-Li tubular struts stiffen the structure for gimballed thrust vector loads at the end of the run tank aft skirt. Symmetrical Al-Li tubular truss thrust structures are used to transfer the engine thrust loads to the run tank aft skirts.

The run tanks are spaced to allow the maximum distance between engines possible without exceeding the 10 meter diameter limit. This provides a distance of 6 meters between the engine centers which is more than the 5 meter minimum required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 9 degrees with the other engine in the neutral position. The overall length of this configuration from start of intertank adapter to engine exit is 23.5 meters.

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM R-2
Mass Properties

ITEM		WEIGHT IN POUNDS	
• CLUSTER TANKED WEIGHT			82490
– Structure		9620	
– Core Tank Lower Support Structure	4930		
– Run Tank Upper Support Structures (2)	690		
– Run Tanks (2) (Includes Insulation)	2959		
– Run Tank Lower Thrust Structure (2)	1040		
– Feed System		1390	
– Feedlines	600		
– Valves	240		
– Disconnects	260		
– Gimbal Joints	210		
– Line Insulation	80		
– Pressurization System		1630	
– Helium Bottles	1290		
– Supports	260		
– Lines	80		
– Engine Assemblies (2)		51320	
– Engines	31480		
– External Shields	19840		
– Contingency (10%)		960	
– Helium		360	
– Hydrogen Capacity		17200	

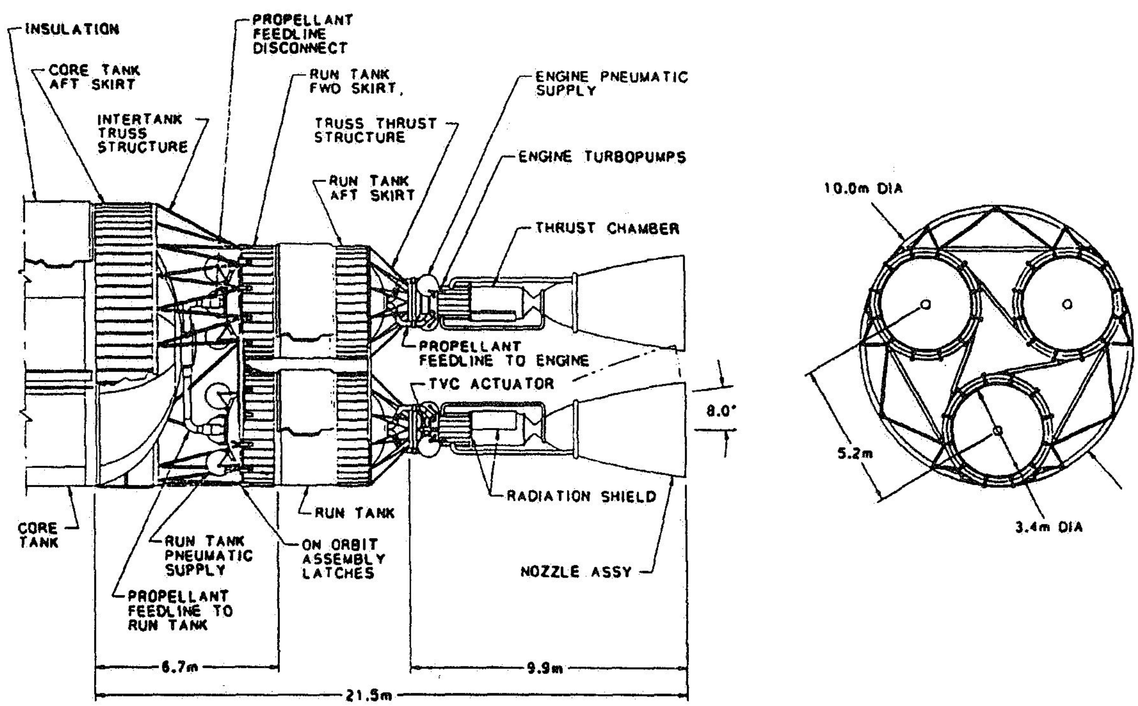
KMS-AFC 1/7/92

NASA LeRC NPO/ASAO reference weights were used for the engine, shield and run tank assemblies. Other weight estimates were developed by General Dynamics Space Systems(GDSS) and are estimated from existing Centaur system weights and/or NLS predesign weights. All structural weights were calculated using Aluminum Lithium (Al 2090, $\rho = 0.092$ lb/in^3). The trussed adapter utilized 24 truss elements per engine. Intertank and run tank adapters were assumed to be semi-monocoque construction. Machined isogrid adapters could be significantly lighter if no frequency/stiffness problems exist. The intertank adapter will likely have many cutouts for fuel lines and/or access. An additional 25% was added to the basic structural weight in order to account for additional localized structure needed around cutouts. An additional 10% contingency factor was added to the GDSS developed weights. The NASA LeRC NPO/ASAO weights were supplied with contingency included.

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM R-3 DESCRIPTION & DIMENSIONS



SHK 1/9/92

This configuration utilizes three 50 klb thrust nuclear thermal rocket engines with separate run tanks. The run tanks are used to minimize pressurization gas requirements for engine start. Gaseous helium for pressurizing the run tanks is supplied by high pressure bottles located above the run tanks. Once engine start is achieved, hydrogen gas is bled from the engines and used to pressurize the core tank. After the core tank is sufficiently pressurized, propellant from the core tank is fed through the run tanks to continue to feed the engines. At the end of each burn, the run tanks may be filled to capacity to repeat the procedure for the next engine start.

The run tank, engine, and thrust structure combine to form the propulsion module. The propulsion module is launched seperately from the rest of the vehicle and is coupled to the core tank on orbit. Fluid system and electrical disconnects and structural latches are provided to allow for on orbit coupling of the propulsion module to the core tank.

An aluminum-lithium tubular intertank truss structure transfers the thrust from the propulsion modules to the core tank. Lateral Al-Li tubular struts stiffen the structure for gimballed thrust vector loads at the end of the run tank aft skirt. Symmetrical Al-Li tubular truss thrust structures are used to transfer the engine thrust loads to the run tank aft skirts.

The run tanks are spaced to allow the maximum distance between engines possible without exceeding the 10 meter diameter limit. This provides a distance of 5.2 meters between the engine centers which is more than the 5 meter minimum required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 8 degrees with the other engine in the neutral position. The overall length of this configuration from start of intertank adapter to engine exit is 21.5 meters.

GENERAL DYNAMICS
Space Systems Division

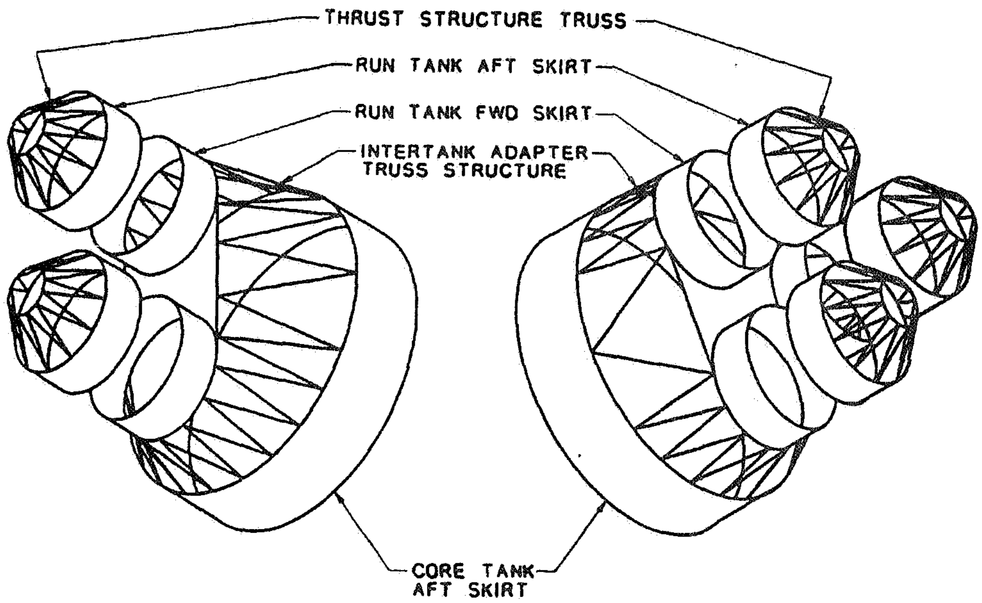
PROPULSION SYSTEM R-3
Mass Properties

ITEM		WEIGHT IN POUNDS	
• CLUSTER TANKED WEIGHT			99000
– Structure		11760	
– Core Tank Lower Support Structure	4850		
– Run Tank Upper Support Structures (3)	970		
– Run Tanks (3) (Includes Insulation)	4440		
– Run Tank Lower Thrust Structure (3)	1500		
– Feed System		1560	
– Feedlines	700		
– Valves	260		
– Disconnects	290		
– Gimbal Joints	230		
– Line Insulation	80		
– Pressurization System		2440	
– Helium Bottles	1930		
– Supports	390		
– Lines	120		
– Engine Assemblies (3)		55780	
– Engines	35940		
– External Shields	19840		
– Contingency (10%)		1130	
– Helium		530	
– Hydrogen Capacity		25800	

KMS-AFC 1/7/92

NASA LeRC NPO/ASAO reference weights were used for the engine, shield and run tank assemblies. Other weight estimates were developed by General Dynamics Space Systems(GDSS) and are estimated from existing Centaur system weights and/or NLS predesign weights. All structural weights were calculated using Aluminum Lithium (Al 2090, $\rho = 0.092$ lb/in^3). The trussed adapter utilized 24 truss elements per engine. Intertank and run tank adapters were assumed to be semi-monocoque construction. Machined isogrid adapters could be significantly lighter if no frequency/stiffness problems exist. The intertank adapter will likely have many cutouts for fuel lines and/or access. An additional 25% was added to the basic structural weight in order to account for additional localized structure needed around cutouts. An additional 10% contingency factor was added to the GDSS developed weights. The NASA LeRC NPO/ASAO weights were supplied with contingency included.

PROPULSION SYSTEM R-2 & R-3
Intertank Adapter / Thrust Structure

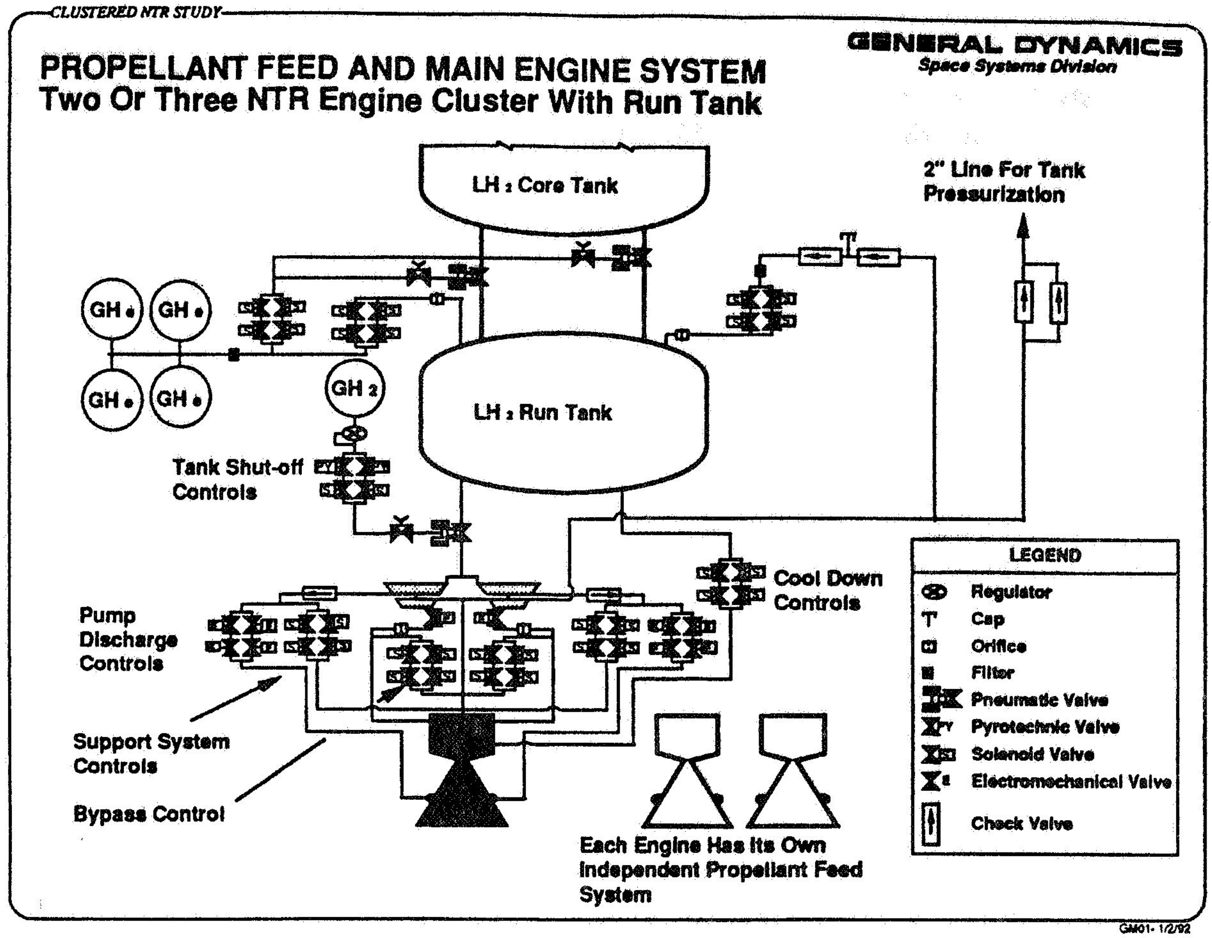


SHK 1/9/92

All the structural components were based on aluminum-lithium construction. Semi-monocoque cylindrical tank skirts were used to be conservative until more stress analysis can be performed Tubular truss structures were used for part of the engine thrust structure and intertank adapter.

The intertank adapter between the core tank and the run tank consists of the core tank aft skirt, the truss structure, and the run tank forward skirt. The sizing of these structures were based on either launch or flight loads. The core tank aft skirt and the truss structure would be launched with the core tank. Launch loads from the fully loaded core tank would be transferred through the aft skirt and into a payload adapter bypassing the truss structure. Launch accelerations were assumed to be 3.0g axial and 1.5g lateral for a 300 klb type heavy lift launch vehicle. The truss structure would only see engine thrust loads once the vehicle was fully assembled. The run tank forward skirt would be launched with the propulsion module on a Titan IV type launch vehicle. The propulsion module would be launched empty and inverted such that the launch loads would be taken through the run tank forward skirt and into the payload adapter. Launch accelerations for a Titan IV type launch vehicle were assumed to be 2.3g axial and 1.5g lateral.

The thrust structure consists of the run tank aft skirt and truss stucture. Both of these structures would also need to withstand the launch loads from a Titan IV type vehicle due to the engine mass since they are all part of the propulsion module.



A propellant feed system with a run tank in addition to the core tank makes it possible to start the propulsion system without pressurizing the core tank first. The smaller volume run tank is pressurized for engine start up. When steady state operation of the engines is established, the core tank is pressurized by autogenous pressurization using hydrogen gas from the turbine outlet. The run tank is then vented enough to allow the tank to be filled with pressurized propellant from the core tank. Two independent main turbopumps were chosen for each engine to guarantee safe engine operation in case of failures in one pump system. The pumps are powered by preheated gaseous hydrogen in an expander cycle arrangement for simplicity and high reliability.

The propellant valves are generally electromechanical. However, due to the large size main propellant feed lines the tank shut-off valves are pneumatically controlled for fast shut-off. The pilot control valves for the pneumatic operated valves are solenoid operated valves. Pyrotechnic valves in the pneumatic system guarantees that the propellant feed system can not be inadvertently opened before the vehicle is ready for operation.

Helium is used for run tank pressurization but an alternative gaseous Hydrogen system could be used with a single 3.5 ft diameter low pressure (300 psia) gas storage bottle that can be continually recharged with hydrogen by feeding liquid Hydrogen from the tank through an electric heater.

Each engine in a two or three engine configuration has its own independent propellant feed system, so that with one engine system out, the mission can be completed with the remaining engine(s).

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

R2, R3 PROPULSION SYSTEMS
Operating Characteristics

	R-2	R-3
START-UP		
THRUST, lbf	0 - 150,000	0 - 150,000
TIME, min	1	1
CORE TANK PRESS., psia	26	26
RUN TANK PRESS., psia	26	26
GIMBAL DISPLACEMENT, degrees	10	6
GIMBAL RATE, degrees/s	5	3
GIMBAL ACCELERATION, degrees/s^2	20	10
STEADY STATE		
THRUST, lbf	56,000 - 150,000	75,000 - 150,000
SPECIFIC IMPULSE, s	925	925
MAXIMUM BURN TIME, min	60	60
CORE TANK PRESS., psia	26 - 40	26 - 40
RUN TANK PRESS., psia	26 - 40	26 - 40
GIMBAL DISPLACEMENT, degrees	10	6
GIMBAL RATE, degrees/s	5	3
GIMBAL ACCELERATION, degrees/s^2	20	10
SHUTDOWN		
THRUST, lbf	150,000 - 380	150,000 - 380
COOLDOWN PULSE FREQUENCY, s^{-1}	∞ - 0.0001	∞ - 0.0001
CORE TANK PRESS., psia	26 - 40	26 - 40
RUN TANK PRESS., psia	26 - 40	26 - 40
GIMBAL DISPLACEMENT, degrees	10	6
GIMBAL RATE, degrees/s	5	3
GIMBAL ACCELERATION, degrees/s^2	20	10

PHS 1/8/92

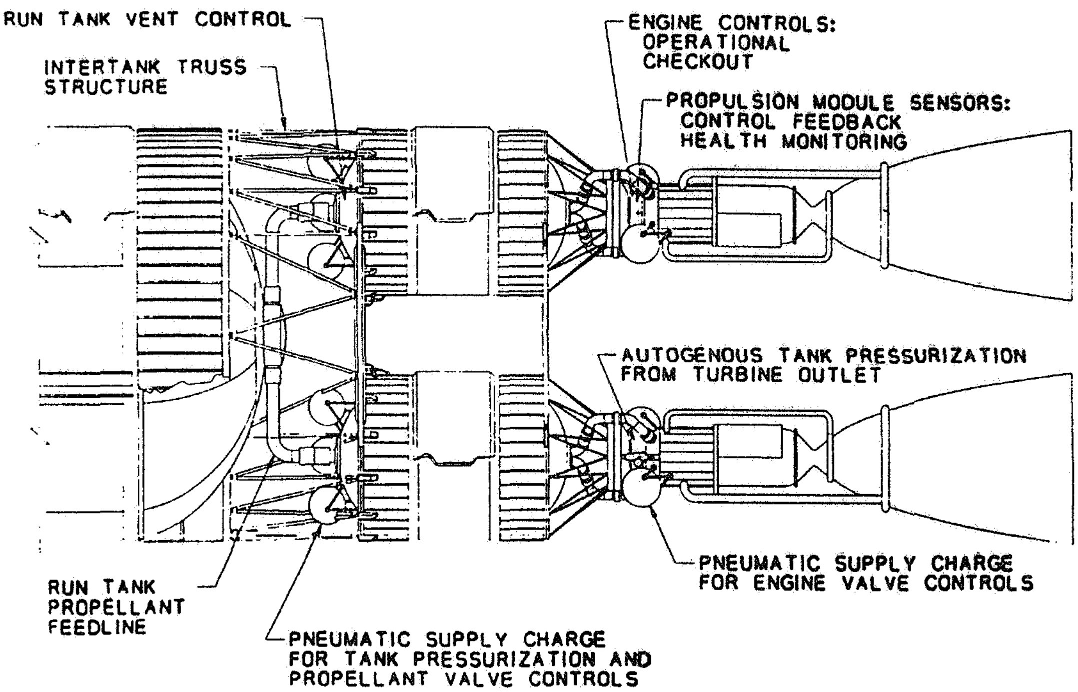
Propulsion system operating characteristics were established for the run tank designs for start-up, steady state, and shutdown conditions. For both the 2-engine and 3-engine designs, the total thrust ramps up from 0 to 150,000 pounds in about 1 minute. Assuming a propellant condition of saturation at 16 psia, about 10 psi pressurization is required to provide NPSH to the engine turbopumps and to account for line entrance losses, line losses, and nuclear radiation heating of the propellant during line transit. The gimbal angular dispacements, slew rates, and accelerations were estimated by adding 2 degrees displacement to the gimbal requirements determined for engine-out events, assuming conditions at the end of start-up.

For steady state, it was assumed that the total thrust could vary from full thrust with all engines operating to an engine-out condition with the active engine(s) throttled to 75 percent thrust. The specific impulse and maximum burn times were assumed to be unchanged from current specifications. For the planned mission, it was estimated that the propellant vapor pressure would rise approximately 14 psi due to nuclear radiation heating of the propellant. The gimbal requirements are the same as at the end of start-up.

The shutdown thrust reduces to a minimum of 190 pounds for the 75,000 lbf NERVA engines. It was estimated that this minimum requirement would scale linearly for the 50,000 lbf engine. The cooldown pulse rate will vary from steady flow to the frequency required at that condition at the point cooling can be terminated (0.0001). The tank pressures and gimballing requirements at the start of shutdown would be the same as for steady state.

GENERAL DYNAMICS
Space Systems Division

INTERFACE ELEMENTS FOR RUN TANK DESIGNS



GM04-1/2/92

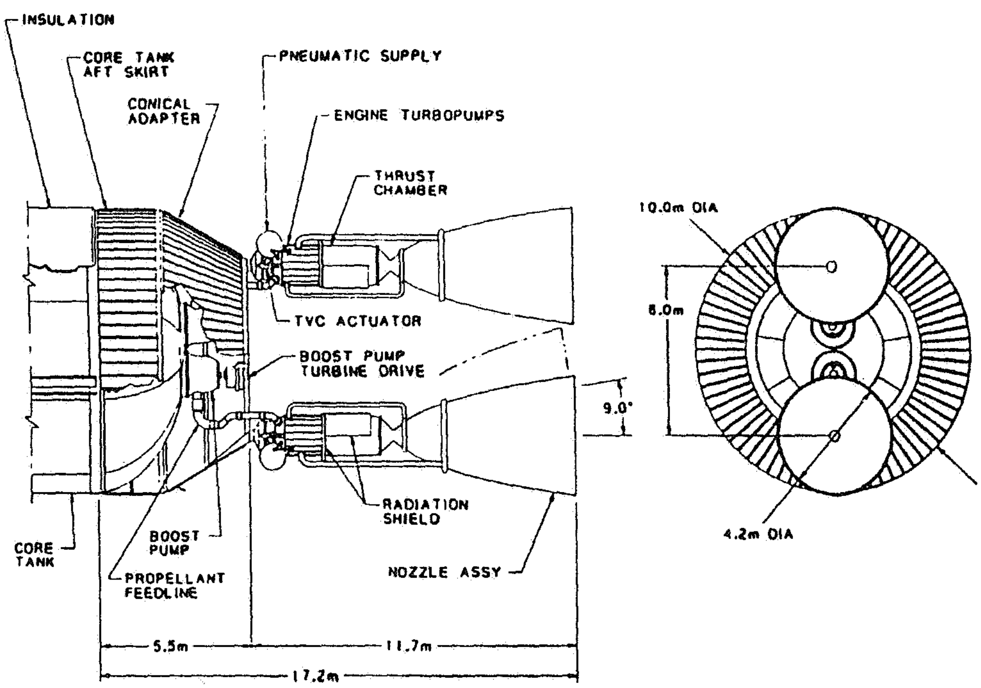
The NTR engine interacts with the vehicle and its support systems through the: engine controllers, engine sensors, control feedback loops, vehicle health managment systems, thrust structure etc. Each of these interface elements is affected in both design and operation by the propulsion system configuration. The thrust structure for example, is sensitive to propulsion system configuration (Run Tanks vs Boost Pumps, etc.) which affect its design on the ground (for access during integration assembly and checkout) and on orbit (depending on assembly philosophy, assembled vs docked vs modular propulsion system design). The other major consideration in the system interface impacts unique to NTR engine based propulsion systems is the radiation field. The propellant feedlines for example are affected by engine in the traditional manner, but with NTR one must also account for operation in an intense radiation environment(propellant heating in lines). Each of the primary interface elements are subject to optimization to minimize mass while maximizing safety and reliability. These systems together have a significant impact on the vehicles performance and design approach and should be integrated into any propulsion system design effort.

BOOST PUMP BASED SYSTEMS

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM B-2 DESCRIPTION & DIMENSIONS



SHK 1/9/92

This configuration utilizes two 75 klb thrust nuclear thermal rocket engines hard-coupled to the core tank. Because of the large ullage volume in the core tank upon restart on some missions, an inordinate amount of pressurization gas would be required to supply the turbopump NPSH for engine restart. Accordingly, the propellant in the core tank is allowed to remain at saturated conditions and boost pumps are used to supply the pressure differential required to provide the NPSH and accomodate the entrance and line losses, as well as the nuclear radiation heating of the propellant as it flows through the line. The boost pumps are powered by turbine drives which run on pressurized gas. Once engine start is achieved, hydrogen gas is bled from the engines and used to run the boost pumps. At the end of each burn the pressurization bottles will be refilled to repeat the procedure for the next engine start.

The core tank, engines, and thrust structure form one unit and are launched together. Due to the fact that this is one unit, the core tank will be shortened by approximately 11.7 meters to accomodate the engines. Extendable nozzles would minimize the launch vehicle shroud volume losses for this configuration. The engine spacing used for this configuration was the same as determined for the run tank version. This provides a distance of 6 meters between the engine centers which is more than the 5 meter minimum required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 9 degrees with the other engine in the neutral position. The overall length of this configuration from start of thrust structure to engine exit is 17.2 meters.

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM B-2
Mass Properties

ITEM	WEIGHT IN POUNDS		
• CLUSTER TANKED WEIGHT			61230
– Structure		6850	
– Core Tank Cylindrical Adapter Structure	3650		
– Conical Thrust Structure	3200		
– Feed System		1450	
– Feedlines	240		
– Valves	240		
– Manifolds	30		
– Gimbal Joints	210		
– Line Insulation	30		
– Boost Pumps	700		
– Helium System		580	
– Helium Bottles	480		
– Supports	90		
– Lines	10		
– Engine Assemblies (2)		51320	
– Engines	31480		
– External Shields	19840		
– Contingency (10%)		890	
– Helium		140	

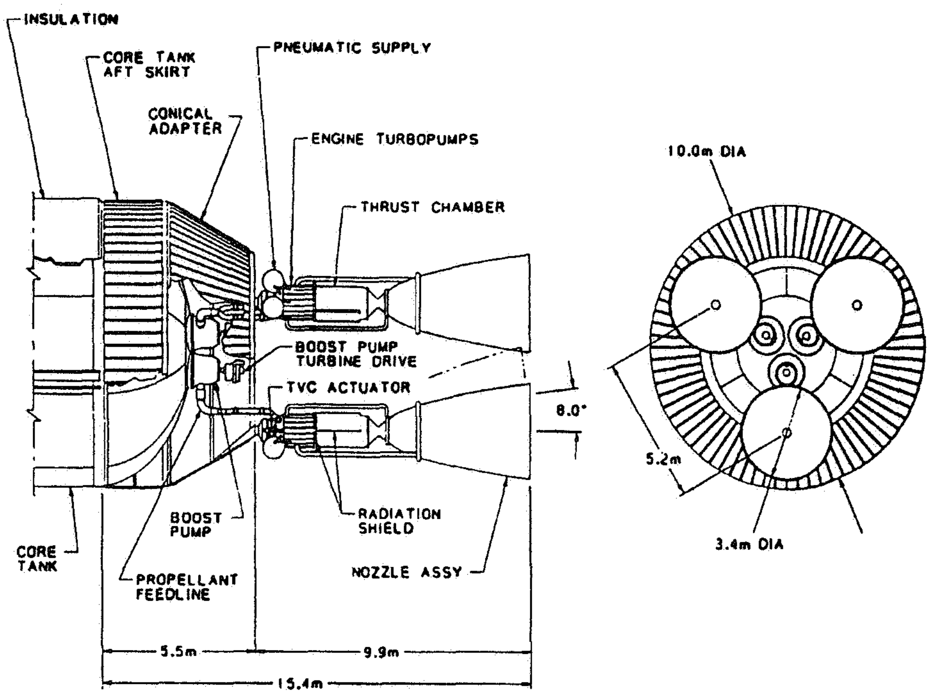
AFC 1/8/92

NASA LeRC NPO/ASAO reference weights were used for the engine and shield assemblies. Other weight estimates were developed by General Dynamics Space Systems(GDSS) and are estimated from existing Centaur system weights and/or NLS predesign weights. All structural weights were calculated using Aluminum Lithium (Al 2090, ρ = 0.092 lb/in^3). Intertank adapters were assumed to be semi-monocoque construction. Machined isogrid adapters could be significantly lighter if no frequency/stiffness problems exist. The intertank adapter will likely have many cutouts for fuel lines and/or access. An additional 25% was added to the basic structural weight in order to account for additional localized structure needed around cutouts. An additional 10% contingency factor was added to the GDSS developed weights. The NASA LeRC NPO/ASAO weights were supplied with contingency included.

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM B-3 DESCRIPTION & DIMENSIONS



SHK 1/9/92

This configuration utilizes three 50 klb thrust nuclear thermal rocket engines hard-coupled to the core tank. Because of the large ullage volume in the core tank upon restart on some missions, an inordinate amount of pressurization gas would be required to supply the turbopump NPSH for engine restart. Accordingly, the propellant in the core tank is allowed to remain at saturated conditions and boost pumps are used to supply the pressure differential required to provide the NPSH and accomodate the entrance and line losses, as well as the nuclear radiation heating of the propellant as it flows through the line. The boost pumps are powered by turbine drives which run on pressurized gas. Once engine start is achieved, hydrogen gas is bled from the engines and used to run the boost pumps. At the end of each burn the pressurization bottles will be refilled to repeat the procedure for the next engine start.

The core tank, engines, and thrust structure form one unit and are launched together. Due to the fact that this is one unit, the core tank will be shortened by approximately 9.9 meters to accomodate the engines. The engine spacing used for this configuration was the same as determined for the run tank version. This provides a distance of 5.2 meters between the engine centers which is more than the 5 meters required to minimize neutronic coupling impacts. This spacing allows one engine to gimbal inboard a maximum of 8 degrees with the other engine in the neutral position. The overall length of this configuration from start of thrust structure to engine exit is 15.4 meters.

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM B-3
Mass Properties

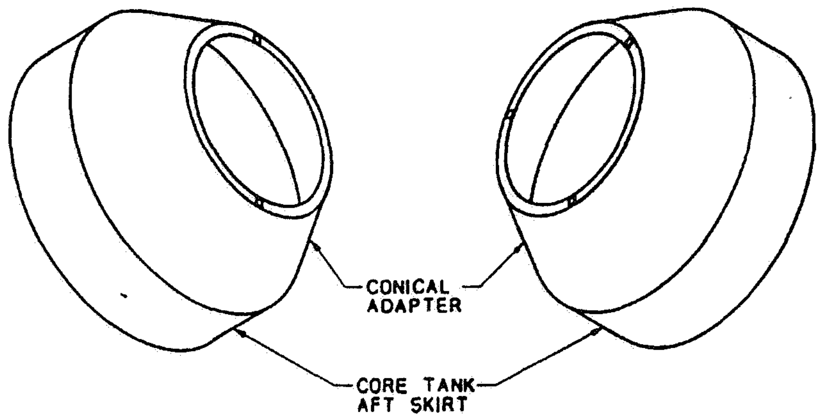
ITEM	WEIGHT IN POUNDS		
• CLUSTER TANKED WEIGHT			66620
– Structure		6850	
– Core Tank Cylindrical Adapter Structure	3650		
– Conical Thrust Structure	3200		
– Feed System		1910	
– Feedlines	470		
– Valves	260		
– Manifolds	20		
– Gimbal Joints	230		
– Line Insulation	30		
– Boost Pumps	900		
– Helium System		900	
– Helium Bottles	730		
– Supports	150		
– Lines	20		
– Engine Assemblies (3)		55780	
– Engines	35940		
– External Shields	19840		
– Contingency (10%)		970	
– Helium		210	

AFC 1/8/92

NASA LeRC NPO/ASAO reference weights were used for the engine and shield assemblies. Other weight estimates were developed by General Dynamics Space Systems(GDSS) and are estimated from existing Centaur system weights and/or NLS predesign weights. All structural weights were calculated using Aluminum Lithium (Al 2090, $\rho = 0.092$ lb/in^3). Intertank adapters were assumed to be semi-monocoque construction. Machined isogrid adapters could be significantly lighter if no frequency/stiffness problems exist. The intertank adapter will likely have many cutouts for fuel lines and/or access. An additional 25% was added to the basic structural weight in order to account for additional localized structure needed around cutouts. An additional 10% contingency factor was added to the GDSS developed weights. The NASA LeRC NPO/ASAO weights were supplied with contingency included.

GENERAL DYNAMICS
Space Systems Division

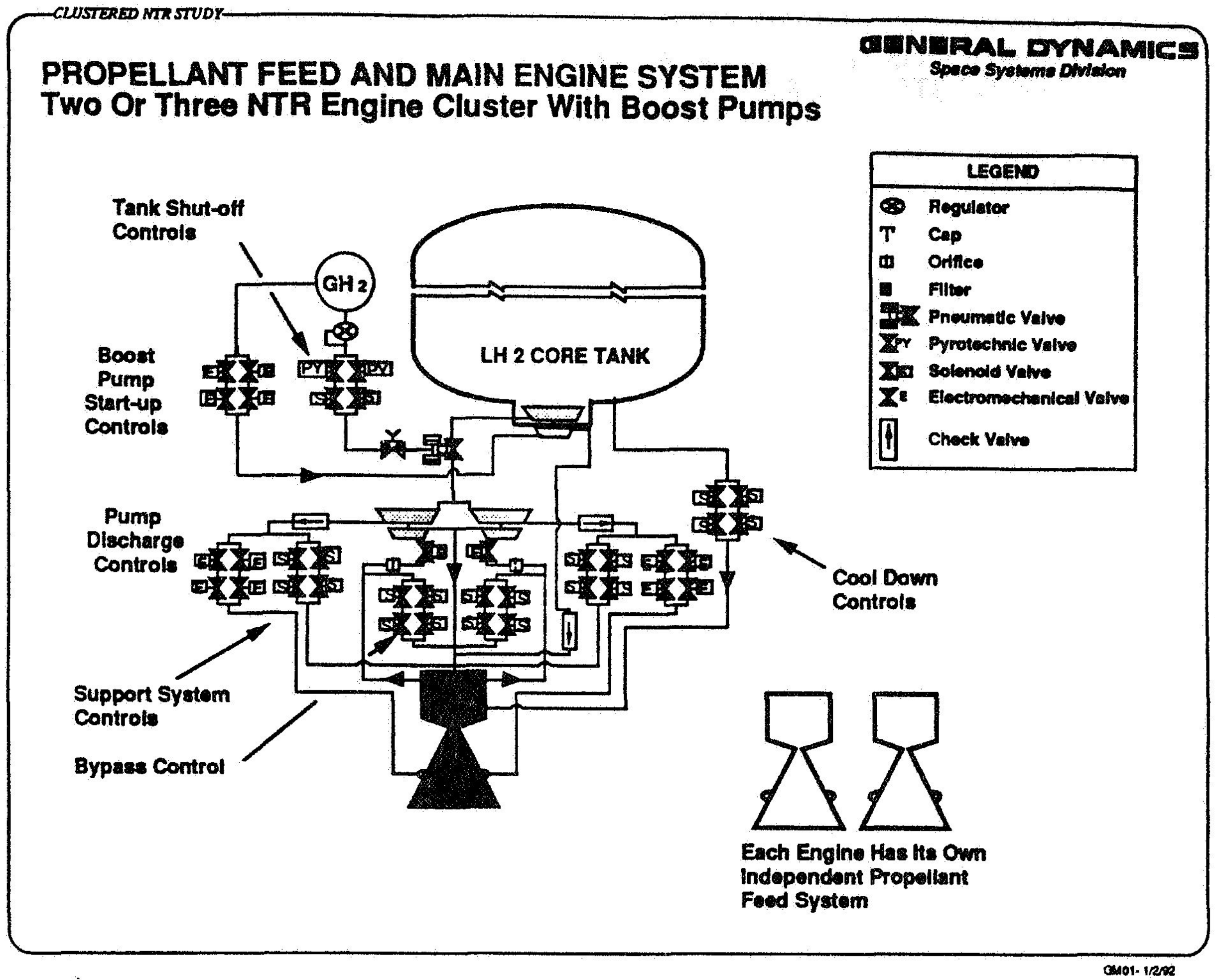
PROPULSION SYSTEM B-2 & B-3 THRUST STRUCTURE



SHK 1/9/92

All the structural components were based on aluminum-lithium construction. To be conservative semi-monocoque structures were used for the core tank aft skirt and conical adapter until more stress analysis can be performed.

The thrust structure consists of the core tank aft skirt and the conical adapter. The sizing of these structures was based on launch loads. For the hard-coupled propulsion system design, the core tank aft skirt, conical adapter, and engines would be launched fully assembled to the core tank. Launch loads from the fully loaded core tank would be transferred through the aft skirt and into a payload adapter bypassing the conical adapter. The conical adapter would have to transfer launch loads from the engine mass into the payload adapter. Launch accelerations were assumed to be 3.0g axial and 1.5g lateral for a 300 klb type heavy lift launch vehicle.



A propellant feed system with boost pumps guarantees a sufficiently high net positive suction head at the propellant inlet to the main engine pumps without tank pressurization. The boost pumps are started by gaseous hydrogen from a storage bottle to initiate rotation of the boostpump turbine drive. Once the engine turbopump head is established, gaseous hydrogen is fed back from the engine cooling jacket outlet to bootstrap the propulsion system propellant head. Two independent main turbopumps were chosen for each engine to guarantee safe engine operation in case of failures in one pump system. The pumps are powered by preheated gaseous hydrogen in an expander cycle arrangement for simplicity and high reliability.

The propellant valves are generally electromechanical. Due to the large size propellant lines and requirements for fast shut-off the main tank shut-off valve is pneumatically controlled.

Gaseous hydrogen is used for the pneumatic control because it can operate with a single 3.5 ft diameter low pressure (300 psia) gas storage bottle that can be continually recharged with hydrogen by feeding liquid hydrogen from the tank through an electric heater or gaseous hydrogen from the engine during engine operation. This bottle can also be used for restart of the boost pump.

Each engine in a two or three engine configuration as shown has its own independent propellant feed system; however, other options are possible.

CLUSTERED NTR STUDY

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PROPULSION SYSTEM OPERATING CHARACTERISTICS - B-2,3

	B-2	B-3
START-UP		
THRUST, lbf	0 - 150,000	0 - 150,000
TIME, min	1	1
CORE TANK PRESS., psia	16	16
GIMBAL DISPLACEMENT, degrees	12	8
GIMBAL RATE, degrees/s	6	4
GIMBAL ACCELERATION, degrees/s^2	25	15
BOOST PUMP DELTA P, psid	10	10
STEADY STATE		
THRUST, lbf	56,000 - 150,000	75,000 - 150,000
SPECIFIC IMPULSE, s	925	925
MAXIMUM BURN TIME, min	60	60
CORE TANK PRESS., psia	16-40	16-40
GIMBAL DISPLACEMENT, degrees	12	8
GIMBAL RATE, degrees/s	6	4
GIMBAL ACCELERATION, degrees/s^2	25	15
BOOST PUMP DELTA P, psid	10	10
SHUTDOWN		
THRUST, lbf	150,000-380	150,000-380
COOLDOWN PULSE FREQUENCY, s^{-1}	∞ - 0.0001	∞ - 0.0001
CORE TANK PRESS., psia	16-40	16-40
GIMBAL DISPLACEMENT, degrees	12	8
GIMBAL RATE, degrees/s	6	4
GIMBAL ACCELERATION, degrees/s^2	25	15
BOOST PUMP DELTA P, psid	10	10

PHS 1/8/92

The propulsion system operating characteristics were established for start-up, steady state, and shutdown conditions. For both the 2-engine and 3-engine designs, the total thrust ramps from 0 to 150,000 pounds in about 1 minute. Assuming a propellant condition of saturation at 16 psia, the boost pump provides an additional 10 psid to provide NPSH to the engine turbopumps and to account for line entrance losses, line losses, and nuclear radiation heating of the propellant during line transit. The gimbal angular dispacements, slew rates, and accelerations were estimated by adding 2 degrees displacement to the gimbal requirements determined for the run tank designs with engine-out events and an additional 2 degrees to account for the reduced displacement of the engines from the c.g. for the boost pump vehicle designs.

For steady state, it was assumed that the total thrust could vary from full thrust with all engines operating to an engine-out condition with the active engine(s) throttled to 75 percent thrust. The specific impulse and maximum burn times were assumed to be unchanged from current specifications. For the planned mission, it was estimated that the shield could be designed to allow the propellant vapor pressure to rise approximately 24 psi due to nuclear radiation heating of the propellant. The gimbal requirements are the same as at the end of start-up.

The shutdown thrust reduces to a minimum of 190 pounds for the 75,000 lbf NERVA engines. It was estimated that this minimum requirement would scale linearly for the 50,000 lbf engine. The cooldown pulse rate will vary from steady flow to the frequency required at that condition at the point cooling can be terminated (0.0001). The tank pressures and gimballing requirements at the start of shutdown would be the same as for steady state.

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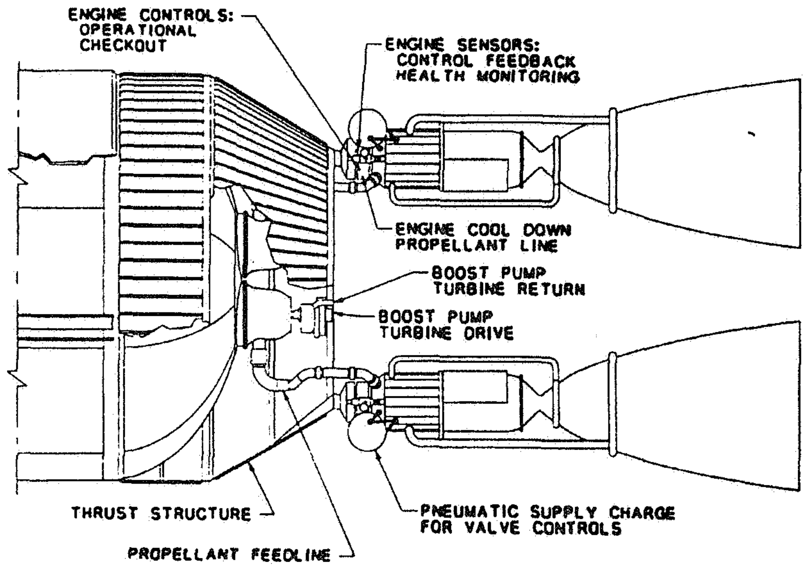
INTERFACE ELEMENTS FOR BOOST PUMP DESIGNS



GM04-1/2/92

The NTR engine interacts with the vehicle and its support systems through the: boost pump turbine drive, boost pump turbine return, engine controllers, engine sensors, control feedback loops, vehicle health managment systems, thrust structure etc. Each of these interface elements is affected in both design and operation by the propulsion system configuration. The thrust structure for example, is sensitive to propulsion system configuration (Run Tanks vs Boost Pumps, etc.) which affect its design on the ground (for access during integration assembly and checkout) and on orbit (depending on assembly philosophy, assembled vs docked vs modular propulsion system design). The other major consideration in the system interface impacts unique to NTR engine based propulsion systems is the radiation field. The propellant feedlines for example are affected by engine in the traditional manner, but with NTR one must also account for operation in an intense radiation environment(propellant heating in lines). Each of the primary interface elements are subject to optimization to minimize mass while maximizing safety and reliability. These systems together have a significant impact on the vehicles performance and design approach and should be integrated into any propulsion system design effort.

CLUSTERED NTR STUDY

GENERAL DYNAMICS
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PROPULSION SYSTEM COMPARISON

	MISSION PERFORMANCE	MISSION RELIABILITY	MISSION OPERATIONS	ENGINE DEVELOPMENT COST	ENGINE DEVELOPMENT SCHEDULE
<u>RUN TANK DESIGN</u>					
2 ENGINE CLUSTER	+				
3 ENGINE CLUSTER		+		+	+
<u>BOOST PUMP DESIGN</u>					
2 ENGINE CLUSTER	+		+		
3 ENGINE CLUSTER	+	+	+	+	+

PHS 1/10/92

A qualitative assessment was made to compare the 4 different engine cluster configurations studied. Some performance advantage can be attributed to the 2-engine cluster designs because the higher thrust engines (75,000 lb) have a somewhat better thrust-to-weight ratio. Also, the boost pump design should be somewhat less weight than the run tank design because less structure is required.

The 3-engine installations provide significantly higher mission reliability, since it would be posssible to continue a mission even after the failure of one engine. The mission would have to be aborted if only 1 engine survives, as would be the case for a 2-engine installation.

The mission operations are simplified with a boost pump, since the engine can be started at any time, whereas the run tanks have to be topped-off before restarting the vehicle with a run tank. Also, the complication of changing over to a core tank supply after start is eliminated with the boost pump design.

The run tank design has somewhat more complicated on-orbit coupling operations due to the necessity of coupling the individual propulsion modules to the aft core tank. With the boost tank design, the engines are hard connected to the aft core tank.

Since the 2-engine cluster designs require higher thrust than the 3-engine designs, the development costs will be higher and the development schedule somewhat longer. A major consideration is the cost of the ground test facilities, which is a function of engine size.

While a weighted scoring was not attempted, it appears that the boost pump design is a somewhat better choice than the run tank design. A 3-engine cluster design appears to be a much better design choice than a 2-engine.

SYSTEMS ANALYSIS

CLUSTERED NTR STUDY

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SYSTEMS ANALYSIS - OBJECTIVES:

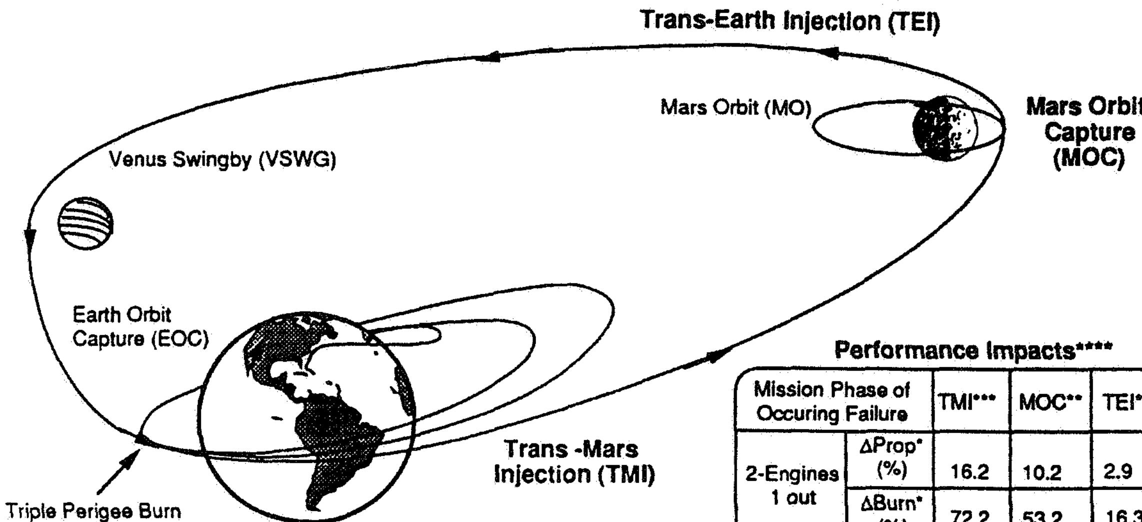
- Identify Engine-Out Impacts on Propulsion Module and Vehicle
- Define/Modify Propulsion System Requirements to Accommodate Engine-Out (RID's)
- Identify Propulsion Module Technology Requirements

JWP 1/8/92

This section examines the impacts of engine out, documents requirements on the engine and vehicle to survive the event, and identifies new propulsion module technology needs.

Our analyses defines mission phases where failures could occur . Thrust vector control requirements to correct for an engine-out condition in both two and three engine configurations are given. Mission performance penalties, in terms of ΔV and additional propellant required to abort, is assessed. Propulsion system failure causes, symptoms, and remedies are examined. The requirements on the surviving propulsion module are defined, and vehicle impacts are discussed. Propulsion module technology requirements are defined, and suggested additions or modifications to the propulsion system baseline are summarized.

ENGINE-OUT MISSION PERFORMANCE IMPACTS



Performance Impacts****

Mission Phase of Occuring Failure		TMI***	MOC**	TEI**
2-Engines 1 out	ΔProp* (%)	16.2	10.2	2.9
	ΔBurn* (%)	72.2	53.2	16.3
3-Engines 1 out	ΔProp* (%)	7.96	4.6	1.3
	ΔBurn* (%)	35.6	24.7	7.7
3 Engines 2 out	ΔProp* (%)	38.5	25.1	6.8
	ΔBurn* (%)	162	123	35.5

- **Analysis assumes failed engine remains with vehicle**
- **Performance impacts diminish as number of engines increase**
- **Gravity Losses included in performance calculations**

**** Values shown are for vehicles with run tank based propulsion systems
*** Assumes failure occurs after escape velocity (C3=0)is achieved
** Assumes engine failure prior to injection or capture maneuvers
* Delta to reference round trip mission parameter for continuing mission after failure occurs

A Mars Transfer System(MTS) with multiple nuclear thermal rocket engines departs from earth parking orbit on 150-day trip to mars. The MTS captures at mars for 90-day stay, then leaves Mars on 310-day return, performing a Venus swingby enroute. The MTS then captures into earth parking orbit.

Main engine failure can occur at any of three mission phases: TMI, MOC, or TEI. During TMI the main engines are utilized for a triple perigee departure burn from Earth parking orbit. Main engines are used again for MOC. The TEI burn is initiated after the 90-day stay on the surface. The failed nuclear thermal rocket engine(s) remains with the vehicle during the entire mission duration.

Our analysis assesses performance impacts due to engine out by calculating *relative* additional propellant mass and reactor burn time delta's to complete the reference mission on time. For the reference mission, total propellant used was approximately 1, 036,000 lbs, while nominal reactor burn time is approximately 1.76 hrs.

The amount of additional propellant required to compensate for engine-out depends on which phase of the mission it occurs. For the TMI phase, full thrust is maintained until escape velocity is reached, then the engine-out condition occurs. For the other two mission phases, engine failure occurs prior the injection or capture maneuvers.

The three engine case affords the least impact for a single engine-out condition. A factor of 2.0 - 2.2 less additional propellant and reactor burn time is required relative to the two-engine case. For two and three engine cases, propellant and burn time impacts are greatly reduced for MOC and TEI failures relative to TMI. This is a result of the reduced gravity well at Mars (.38 of Earth's).

The vehicle with a boost pump based propulsion system was also analyzed. Its Δpropellant and Δburn times were 8% and 3% lower respectively than the run tank case.

CLUSTERED NTR STUDY

PROPULSION SYSTEM FAILURE MODE ANALYSIS

GENERAL DYNAMICS
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NTR Engine System With Runtank	Problem	Impact	Symptoms	Diagnostics	Remedy
• PROPELLANT SUPPLY SYSTEM					
• CONTROL BRANCHES					
• CORE / RUN TANK PROP. CONTROL VLVS.	2 Branches Fail	No Or Uncon-Trolled Flow	Press. Off Nominal	Pressure	Abort
• RUN TANK PRESSURIZATION VLVS.	Low NPSH	Flow Instability	Pressure Fluctuation	Runtank Pressure	Temporary Reduced Startup Thrust
• MAIN PROPELLANT START-UP CTL. VLVS.	2 Branches Fail	No Or Uncon-Trolled Flow	Press. Off Nominal	Pressure	Abort
• RUN TANK VENT CTL. VLVS	Boiling Instability	Flow Instability	Pressure Fluctuation	Runtank Pressure	Reduce Thrust
• GAS STORAGE	Lose Pressure	Abort Mission	Pressure Low	Pressure	Shut Down Engine And Abort
• AUTOGENOUS TANK PRESSURIZATION	Low Npsh	Flow Instability	Pressure Fluctuation	Run-/Coretank Pressure	Reduce Thrust
• THRUST VECTOR CONTROL SYSTEM					
• TVC ACTUATORS (EMA)	Lose Power	Slow Ctl. Response	Parameter Response Vs	Current Degree / Sec	Shut Down One Engine Affected
• CONTROLLER (EMA)	Lose Signal	Lose Control	Command Monitor	Voltage Indicator	Shut Down One Engine Affected
• MAIN ENGINE					
• ENGINE CONTROLLER	Bad Signal	Lose Control	T&P Variation	T and P	Shutdown/Jettison
• CONTROL BRANCHES	2 Branches Fail Closed	Loss Of Flow	Reduced P Or Flow Loss	Valve Position Indicator	Shut Down 1 Pump (Reduce Thrust)
• PUMP DISCHARGE CTL VLVS	Lose Regulation				
• PUMP BYPASS CTL VLVS.	↓	Loss Of Flow	Reduced P	P, Flow., Vlv. Pos.	Shutdown/Jettison
• TURBINE BYPASS CTL VLVS.	↓	Loss of P Ctl.	Reduced P	P, Flow., Vlv. Pos.	Shutdown/Jettison
• COOLDOWN CTL VLVS.	↓	Loss of reactor	Vap. Fuel Elem.	Lo flow, Hi reactor T	Shutdown/Jettison
• TURBOPUMP (ONE OF TWO)	Bearings Or Blades Fail	Reduced Or No Flow	Pressure Loss	Flowmeter Pressure	Shut Down Pump (Eng. Thrust Reduc.)
• THRUST CHAMBER					
• REACTOR	No Coolant	Core Melt Exhausted	Hi Temp.	Temp.	Shutdown/Jettison
• CONTROL DRUM	Fail To Rotate	Lose Power Control	Hi Temp.	Temp.	Apply Back-up Control
• REACTOR PRESSURE VESSEL	Overheat	Fracture	Hi T's And P's	T And P	Shutdown/Jettison
• NOZZLE / GASIFIER HX.	OVERHEAT	FRACTURE	HI T's AND P's	T AND P	Shutdown/Jettison

GM09-08/01/92

The above table summarizes the most serious problems that can occur for a nuclear propulsion system with a run tank; however, although they are also the least likely to occur. Many small failures can occur in the support systems undetected and without having any impact on the operation of the propulsion system because of the redundancy and safety features built into the systems. In most cases reduced thrust or safe abort is possible.

The propulsion system includes not only the main engine hardware, but also integrated support systems containing numerous valves, electrical switches, regulators, high pressure gas storage systems, etc, all of which are carefully chosen for specific functions and arranged in multiple combinations to guarantee safe, reliable and accurately controlled operation of the overall propulsion system. Problems associated with the propulsion system is therefore not only related to the main hardware components but also to the many components of the integrated support systems. Problems and failures in the overall system are most often related to the support systems and are detected by instrumentation and behavior of the support systems.

Problems and failures in NTR systems are related mostly to the systems and components which are similar to conventional chemical rockets, which makes it easier to analyze the NTR systems based on past experience. With failures in the reactor, it may be possible to continue safe operation at reduced power for even extended periods of time, since reactor life is greatly increased at reduced power.

Where practical, electromagnetic valves and actuators were chosen for high reliability and fast response. Electromagnetic hardware has been demonstrated to be better performing than pneumatic or hydraulic systems in many applications. Problem areas are mostly related to the controllers in the system which therefore require a large degree of redundancy built into the control systems.

PROPULSION SYSTEM FAILURE MODE ANALYSIS

GENERAL DYNAMICS
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NTR Engine Syst. With Boost Pump	Problem	Impact	Symptoms	Diagnostics	Remedy
• **PROPELLANT SUPPLY SYSTEM**					
• **CONTROL BRANCHES**					
• BOOST PUMP START-UP CONTROL VALVES	2 Branches Fail	No Or Uncon-Trolled Flow	Pressure Off Nominal	Pressure	Shut Down Pump W. P Out Of Bound
• MAIN PROPELLANT START-UP CONTROL VALVES	Low NPSH	Flow Instability	Pressure Fluctuation	Core Tank Pressure	Reduce Thrust
• **GAS STORAGE**	Lose Pressure	Abort Mission	Pressure Low	Pressure	Shut Down Engine And Abort
• **THRUST VECTOR CONTROL SYSTEM**					
• **TVC ACTUATORS (EMA)**	Lose Power	Slow Control Response	Parameter Response Vs	Current Degree / Sec	Shut Down One Engine Affected
• **CONTROLLER (EMA)**	Lose Signal	Lose Control	Command Monitor	Voltage Indicator	Shut Down One Engine Affected
• **MAIN ENGINE**					
• **ENGINE CONTROLLER**	Bad Signal	Lose Control	T&P Variation	T and P	Shutdown/Jettison
• **CONTROL BRANCHES**	2 Branches Fail Closed	Loss Of Flow	Reduced P Or Flow Loss	Valve Position Indicator	Shut Down 1 Pump (Reduce Thrust)
• PUMP DISCHARGE CTL. VLVS	Lose Regulation				
• PUMP BYPASS CTL. VLVS.	↓	Loss Of Flow	Reduced P	P, Flow., Vlv. Pos.	Shutdown/Jettison
• TURBINE BYPASS CTL. VLVS.	↓	Loss of P Ctl.	Reduced P	P, Flow., Vlv. Pos.	Shutdown/Jettison
• COOLDOWN CTL. VLVS.	↓	Loss of reactor	Vap. Fuel Elem	Lo flow, Hi reactor T	Shutdown/Jettison
• **TURBOPUMP (ONE OF TWO)**	Bearings Or Blades Fail	Reduced Or No Flow	Pressure Loss	Flowmeter Pressure	Shut Down Pump (Eng. Thrust Reduc.)
• **THRUST CHAMBER**					
• REACTOR	No Coolant	Core Melt Exhausted	Hi Temp.	Temp.	Shutdown/Jettison
• CONTROL DRUM	Fail To Rotate	Lose Power Control	Hi Temp.	Temp.	Apply Back-up Control
• REACTOR PRESSURE VESSEL	Overheat	Fracture	HI T's And P's	T And P	Shutdown/Jettison
• NOZZLE / GASIFIER HX.	Overheat	Fracture	HI T's AND P's	T AND P	Shutdown/Jettison

GM08-08/01/92

The above table summarizes the most serious problems that can occur for a nuclear propulsion system with a boost pump; however,they are also the least likely to occur. Many small failures can occur in the support systems undetected and without having any impact on the operation of the propulsion system because of the redundancy and safety features built into the systems. In most cases reduced thrust or safe abort is possible.

The propulsion system includes not only the main engine hardware, but also integrated support systems containing numerous valves, electrical switches, regulators, high pressure gas storage systems, etc, all of which are carefully chosen for specific functions and arranged in multiple combinations to guarantee safe, reliable and accurately controlled operation of the overall propulsion system. Problems associated with the propulsion system is therefore not only related to the main hardware components but also to the many components of the integrated support systems. Problems and failures in the overall system are most often related to the support systems and are detected by instrumentation and behavior of the support systems.

Problems and failures in NTR systems are related mostly to the systems and components which are similar to conventional chemical rockets, which makes it easier to analyze the NTR systems based on past experience. With failures in the reactor, it may be possible to continue safe operation at reduced power for even extended periods of time, since reactor life is greatly increased at reduced power.

Where practical, electromagnetic valves and actuators were chosen for high reliability and fast response. Electromagnetic hardware has been demonstrated to be better performing than pneumatic or hydraulic systems in many applications. Problem areas are mostly related to the controllers in the system which therefore require a large degree of redundancy built into the control systems.

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THRUST VECTOR CONTROL REQUIREMENTS

- FLIGHT PATH STEERING
- VEHICLE C.G. OFFSET
- THRUST DIFFERENTIAL
- NON-UNIFORM DEPLETION OF PROPELLANTS
- ENGINE-OUT
- TANK JETTISONING
- ENGINE JETTISONING
- PROPELLANT SLOSHING
- VEHICLE ELASTIC MOTION

PHS 1/8/92

Various vehicle factors have to be considered in determining the thrust vector control requirements for a space vehicle.

A basic consideration is the flight path steering requirement. For an orbit launched vehicle, this requirement is minimaland is not critical .

A number of alignment factors, including thrust differential, vehicle c.g. offset, non-uniform depletion of propellant, engine-out, tank jettisoning, and engine jettisoning, require adjustment of the thrust vector.

Propellant sloshing and vehicle elastic motion may be coupled and must be considered in that context.

While a comprehensive survey has not been accomplished for this study, it was recognized that the engine-out event could have a major impact on the requirements. Accordingly, an assessment of this particular factor was made to obtain an indication of the magnitude of the requirement.

SURVIVING PROPULSION MODULE REQUIREMENTS

- Reactor burn time requirements could increase by 7% to 35% for the three engine case and 16% to 70% for the two engine case
- If no engine jettison capability incorporated in design, surviving propulsion modules must be able to function in intense radiation and thermal environment caused by disabled engine
- Gimbal requirements for run tank designs are worst for 1 of 2 engines out at maximum displacement, rate and accelleration of 7.5°, 3.5°/sec and 20°/sec^2 respectively
- Gimbal mechanism must be robust and capable of vehicle control for extended duration at or near the maximum engine out null position of 5°

KMS 1/17/92

THRUST VECTOR CONTROL REQUIREMENTS

- FLIGHT PATH STEERING
- VEHICLE C.G. OFFSET
- THRUST DIFFERENTIAL
- NON-UNIFORM DEPLETION OF PROPELLANTS
- ENGINE-OUT
- TANK JETTISONING
- ENGINE JETTISONING
- PROPELLANT SLOSHING
- VEHICLE ELASTIC MOTION

PHS 1/8/92

Various vehicle factors have to be considered in determining the thrust vector control requirements for a space vehicle.

A basic consideration is the flight path steering requirement. For an orbit launched vehicle, this requirement is minimaland is not critical (unless a meteor/debris avoidance system is included).

A number of alignment factors, including thrust differential, vehicle c.g. offset, non-uniform depletion of propellant, engine-out, tank jettisoning, and engine jettisoning, require adjustment of the thrust vector.

Propellant sloshing and vehicle elastic motion may be coupled and must be considered in that context.

While a comprehensive survey has not been accomplished for this study, it was recognized that the engine-out event could have a major impact on the requirements. Accordingly, an assessment of this particular factor was made to obtain an indication of the magnitude of the requirement.

GENERAL DYNAMICS
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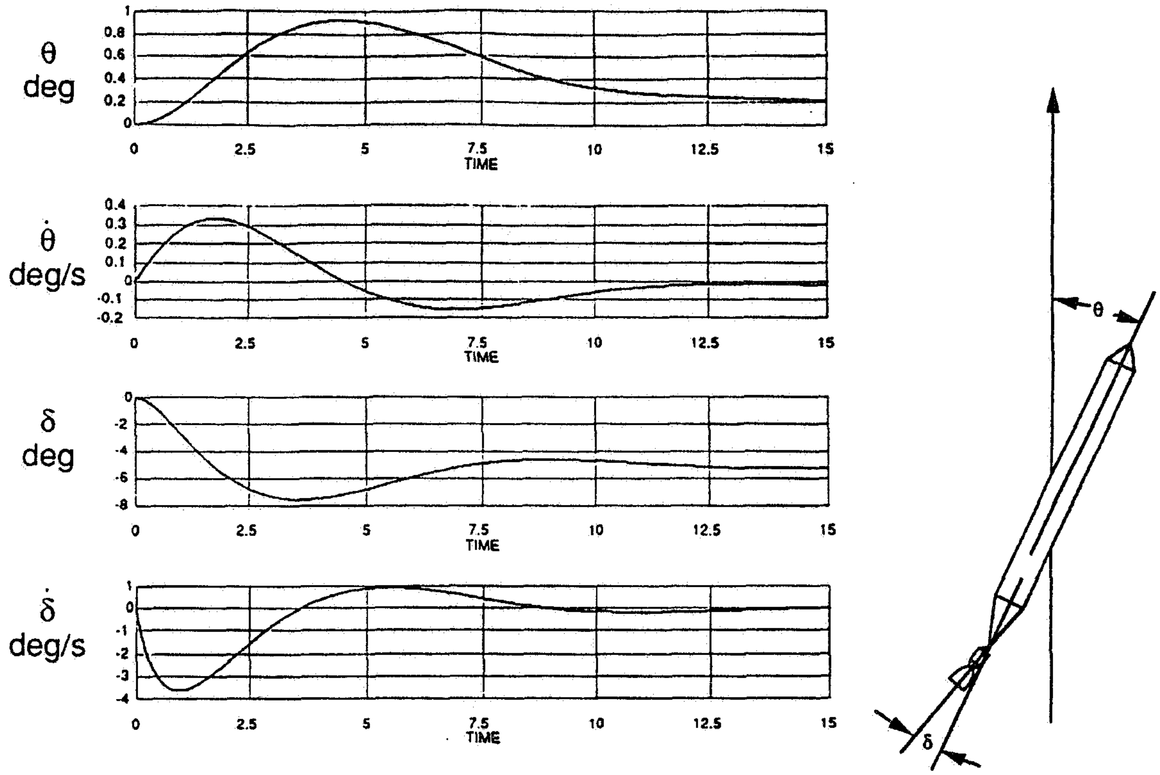
THRUST VECTOR CONTROL REQUIREMENTS - 1 of 2 Engines Out



PHS 1/8/92

A dynamic guidance and control simulation was developed to examine the vehicle control response and engine gimballing requirements for the case where one engine of 2 fails. The vehicle mass distribution was examined and it was concluded that the worst case could be approximated with the Mars transfer injection maneuver tanks jettisoned and the aft core tank full of propellant. Instantaneous shutdown of the faulted engine also was assumed for the worst case.

A control loop was formulated and typical PID (Proportional Integral Differential) control gains were applied to obtain what appeared to be favorable results. As indicated in the plots of the results, the maximum vehicle alignment excursion is less than 1 degree (occurring about 4.5 seconds into the transient) and the excursion rate is about 0.3 degrees/second (at 2 seconds after engine thrust termination). The maximum engine gimbal response is approximately 7.5 degrees (at 3.5 seconds) requiring a maximum gimbal rate of about 3.5 degrees/second (at about 1 second). The engine-out null position is about 5 degrees parallel to the radial position vector of the faulted engine.

The control simulation maximum gimbal acceleration is about 20 degrees/second^2. This compares with a total deflection rate of 114 degrees/second^2 for the Centaur engines.

CLUSTERED NTR STUDY

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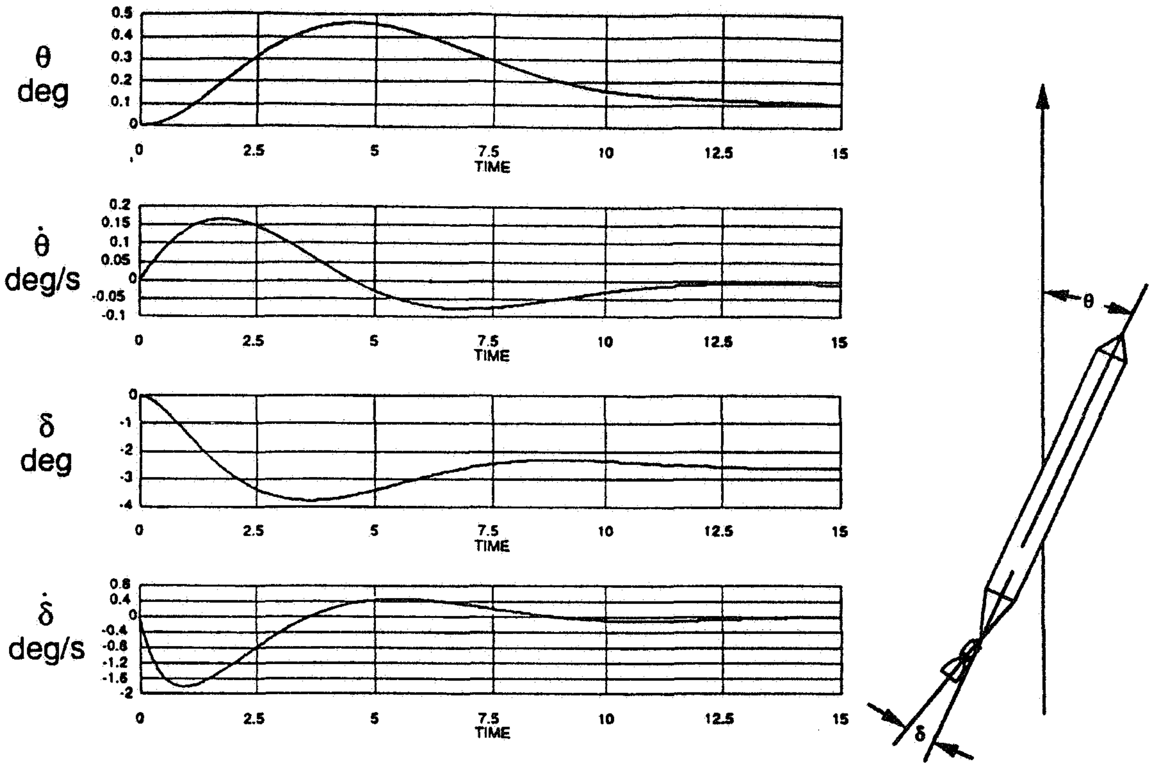
THRUST VECTOR CONTROL REQUIREMENTS - 1 of 3 Engines Out



PHS 1/8/92

A dynamic guidance and control simulation was developed to examine the vehicle control response and engine gimballing requirements for the case where one engine of 3 fails. The vehicle mass distribution was examined and it was concluded that the worst case could be approximated with the Mars transfer injection maneuver tanks jettisoned and the aft core tank full of propellant. Instantaneous shutdown of the faulted engine also was assumed for the worst case.

A control loop was formulated and typical PID (Proportional Integral Differential) control gains were applied to obtain what appeared to be favorable results. As indicated in the plots of the results, the maximum vehicle alignment excursion is less than 0.5 degree (occurring about 4.5 seconds into the transient) and the excursion rate is less than 0.2 degrees/second (at 2 seconds after engine thrust termination). The maximum engine gimbal response is approximately 4 degrees (at 3.5 seconds) requiring a maximum gimbal rate of 2 degrees/second (at about 1 second). The engine-out null position is about 3 degrees parallel to the radial position vector of the faulted engine.

The control simulation maximum gimbal acceleration is about 10 degrees/second^2. This compares with a total deflection rate of 114 degrees/second^2 for the Centaur engines.

CONCLUSIONS - THRUST VECTOR CONTROL REQUIREMENTS

- ENGINE-OUT REQUIREMENTS:

	1 OF 2	1 OF 3
DISPLACEMENT, degrees	7.5	4
RATE, degrees/second	3.5	2
ACCELLERATION, degrees/s^2	20	10
NULL, degrees	5	3

- THE ENGINE-OUT REQUIREMENTS FOR THE BOOST PUMP DESIGNS WILL BE GREATER AND SHOULD BE ANALYZED.
- TANK AND ENGINE JETTISON CONDITIONS COULD BE SIGNIFICANT AND SHOULD BE ANALYZED.
- ALLOWANCE OF ABOUT 2 DEGREES APPEARS TO BE ADEQUATE FOR OTHER REQUIREMENTS.

PHS 1/8/92

The thrust vector control requirements have been determined for the engine-out event for the vehicle designed with a run tank. The displacement, gimbal rate, gimbal acceleration, and null position are about twice as great for the 3 engine installation with the run tank design as they are for the 2 engine installation of the same basic design. For the designs with the run tank, the requirements do not appear to be excessive. The requirements for the boost pump propulsion system designs could be significantly greater, however, and the should be analyzed.

The requirements for tank and engine jettison events could also be significant and should be analyzed. In any case, however, these events would not be concurrent with the engine-out event and therefore should not increase the overall gimballing requirements.

Some other conditions and events, such as propellant sloshing, could be concurrent with engine-out. Such additional requirements should not be major, however, and probably can be covered with a nominal allowance of, say, 2 degrees.

TECHNOLOGY REQUIREMENTS

GENERAL DYNAMICS
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1) Robotic Coupling Tools/Techniques - On-orbit Assembly Of Core Tanks & Propulsion Modules

2) On-orbit Propellant Transfer - Top-off Propellant Tanks For Maximum Capability Missions

3) Boost Pumps - Six Times Mass Flow Of Centaur, GH2 Turbine Drive

4) Radiation Hardened Thrust Vector Controlers And Engine Controllers - Gamma Heating And Charged Particle Upsets

5) Run Tank Vent/Fill Systems - Vent GHe From Run Tank For In-space Restarts

6) Mixing Conditions With Bulk Heated Propellant - Predict LH2 Temperature For Turbopump Restart Conditions

7) Engine Jettison System - Reduce Engine-out Vehicle Mass, Prevent Good Engine Obstruction By Dysfunctional Engine

8) Integrated Health Monitoring & Built-In Test - Reduce In-space Checkout Time/ Cost, Automatically Compensate For failed/off-nominal Conditions

JWP 12/19/91

Robotic coupling tools/ techniques refers to the need for an OMV-type tug with mechanical manipulation capbilities. This tool would be used to mate core tanks, auxilliary tanks, propulsion modules and other components after delivery by launch vehicle to an assembly orbit.
On-orbit propellant transfer capability would be needed to compensate for boiloff during assembly periods, and undertanking done to keep within launch vehicle delivery constraints.
Boost pumps were developed for Centaur. However, those needed for an NTR stage would need to maintain six times their mass flow. Centaur turbopumps were driven by Hydrogen Peroxide. Those for an NTR stage might require special materials since driven by GH2 from the engine cooling jacket.
Radiation hardened controllers for the engine and fluid system valves will be required Normal Centaur electronics would be subject to gamma heating damage and charged particle upset.
Run tank vent/fill systems must be developed. At mission start, the run tank is pressurized with GHe. After MECO 1, propellant from core tanks is to fill and flow thru the run tanks to the engines. For minimum ullage, pressurization gases need to be vented from the run tank before it receives with the core propellant.
Mixing conditions with bulk heated propellant need to be examined with CFD codes and experimental simulators able to model and predict existance of either stratified or circulation mixed propellant conditions. Unlike circulation mixed fluid, stratified propellant could allow non-uniform temperature of LH2 delivered to turbopump inlets over the duration of tank drain. Additional tank pressurization, which could impact design, would be needed to ensure net positive suction head conditons were met at all times.
An engine jettison system to discard a disfunctional propulsion module would reduce the mass of the vehicle, and thus the performance impact of engine-out. It would eliminate the possibility of a dead module staying critical and physically distorting to obstruct gimballing the remaining module.
IHM and Smart Bit allow automated checkout to reduce the cost of in-space operations. It also allows off- nominal and inipient failure conditions to be compensated for at electronic speed.

SUMMARY AND CONCLUSIONS

GENERAL DYNAMICS
Space Systems Division

CONCLUSIONS

- A cluster of multiple propulsion modules coupled to a core tank is feasible
- Hard coupling multiple nuclear thermal rocket engines to a core tank is an attractive alternative
  - Boost pumps utilized for engine start/restart
  - Upper and lower core tanks
  - Propulsion system integrated and checked out on the ground for single launch on an HLLV
- Three engine cluster appears to be more desirable than two engine cluster
  - Higher reliability
  - Less performance penalty in engine out scenario
  - Reduced reactor burn time requirements in engine out scenario
  - Lower thrust engines may cost less to develop
- May not be desirable to abort with 1 of 3 engines out, after TMI
  - Minimal reactor and propellant penalties, great mission success benefits

GENERAL DYNAMICS
Space Systems Division

PROPULSION SYSTEM ISSUES

- NUMBER/THRUST OF ENGINES
- ABORT STRATEGY VS. NUMBER OF ENGINES
- SINGLE VS. DUAL TURBOPUMPS - F(NO. OF ENGINES)
- NPSH SUPPLY CONCEPT (RUN TANK VS. BOOST PUMP)
- ENGINE JETTISON PROVISIONS
- RADIATION HARDENING OF ENGINE/TVC CONTROLLERS
- FEASIBILITY OF VARIABLE INTERNAL SHIELD (VS. SPLIT SHIELD)
- ROBOTIC REQUIREMENTS FOR ON-ORBIT COUPLING
- ON-ORBIT PROPELLANT TRANSFER REQUIREMENTS
- ENGINE CHECKOUT REQUIREMENTS
- ENGINE HEALTH MONITORING REQUIREMENTS

PHS 1/9/92

A number of major issues have to be resolves in order to adequately specify the propulsion sytem for a nuclear space transfer vehicle. The number and thrust of the engines must be determined based on an analysis of the initial and ultimate mission requirements, the cost of ground test facilities, and launch manifest considerations, as well as the reliability and engine-out capablilty of the engine cluster. The vehicle abort strategy and resolution of the requirement for single vs. dual turbopumps, in turn, are dependent upon the decision on the number of engines in the cluster.

The necessity for reactor cooling of a faulted engine can be avoided and abort mission performance can be improved if the faulted engine can be jettisoned. The experience available from the launch of 500 Atlases with jettisoned booster engines should be applied to determine what jettison features can be used with the clustered nuclear rocket propulsion system.

A major weight savings can be achieved if the engine thrust chamber can be developed with a variable internal shield. This is particularly critical with a clustered engine installation where side shielding is required to protect the adjacent run tank (or far side of the core tank bottom, in the case of the design with a boost pump). Accordingly, the feasibility of designing the thrust chamber with a variable internal shield should be explored.

On-orbit issues include coupling of the propulsion module to the vehicle and propellant transfer. The requirements for these operations should be analyzed in the context of design and development implications.

In order to establish the number and type of control and sensor interface connectors that must be provided, a definition of the engine checkout and health monitoring requirements must be derived. These requirements could have a major impact on the concept and location of the connector panels used for coupling the propulsion module to the vehicle.

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

SUMMARY AND CONCLUSIONS

- Engine jettison capability is a must
  - Jettisoning failed engine will improve reactor and propellant abort margins as well as reduce radiation and thermal protection requirements on the surviving propulsion module(s)
- Radiation hardening of engine/TVC controllers should result in substantial weight savings
  - The alternative is local electronics shielding or side shields on reactor
  - Ability to harden/locally shield will drive selection of TVC actuator
- Side shielding of reactor may offer substantial design/operations benefits
  - Reduce disk shield mass
  - Simpler installation on ground or in orbit
- TVC actuator displacement and gimbal rate and power requirements are within current state of the art
  - Displacement and rate calculated, actuator power by analogy

KMS 1/7/92

CLUSTERED NTR STUDY

GENERAL DYNAMICS
Space Systems Division

SUMMARY AND CONCLUSIONS

- Propulsion system design is dependant on man-rating requirements
- On orbit propellant transfer for tanking/topping propellant tanks
- Run tanks should be launched empty
  - Large surface area/volume ratio and on orbit assembly time
  - Reduce structural mass requirements
- Integrated health management is a must for any NTR
  - IHM/Smart Bit architecture implications will have significant impact on propulsion system mass and reliability
- The Earth to Orbit lift and volume constraints, coupled with on orbit operation significantly affect the propulsion module/system design

KMS 1/7/92

N93-26926

NASA LEWIS RESEARCH CENTER

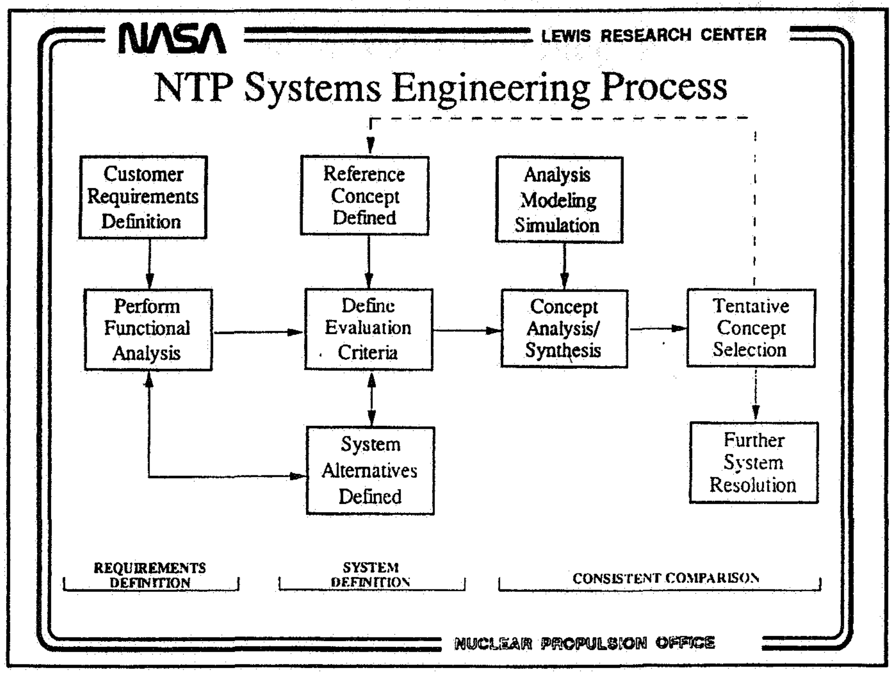
NTP Comparison Process

Nuclear Propulsion Technical Interchange Meeting

Sandusky, OH
October 2,1992

Robert Corban
Nuclear Propulsion Office
NASA Lewis Research Center

NUCLEAR PROPULSION OFFICE



NTP Systems Engineering Process

The systems engineering process is shown above is for the concept definition phase of the program. The process involves three major elements: requirements definition, system definition, and consistent concept comparison. The requirements definition process involves obtaining a complete understanding of the system requirements based on customer needs, mission scenarios, and NTP operating characteristics. A system functional analysis is performed to provide a comprehensive traceability and verification of top-level requirements down to detailed system specifications and provides significant insight into the measures of system effectiveness to be utilized in system evaluation. The second key element in the process is the definition of system concepts to meet the requirements. This part of the process involves engine system and reactor contractor teams to develop alternative NTP system concepts that can be evaluated against specific attributes, as well as a reference configuration against which to compare system benefits and merits. Establishing the evaluation criteria will be extremely challenging and critical to the entire evaluation and selection process. Due to the various disciplines required and many goals the system will be required to achieve, an iterative and participative team approach must be utilized. Various methodologies exist for evaluating a comprehensive set of evaluation criteria: analytic hierarchy process (AHP), multiple-attribute-utility method (MAUM), and weighted-outranking method (WOM), but these provide little structure in identifying the key criteria. Quality function deployment (QFD), as an excellent tool within Total Quality Management (TQM) techniques, can provide the required structure and provide a link to the "voice" of the customer in establishing critical system qualities and their relationships. The third element of the process is the consistent performance comparison. The comparison process involves validating developed concept data and quantifying system merits through analysis, computer modeling, simulation, and, if required, rapid prototyping of the proposed high risk NTP subsystems. The maximum amount possible of quantitative data will be developed and/or validated to be utilized in the QFD evaluation matrix. If upon evaluation of a new concept or its associated subsystems determine to have substantial merit, those features will be incorporated into the reference configuration for subsequent system definition and comparison efforts.

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Requirements Definition

- **Customers**
  - Understand who are our customers
  - Need to understand customer needs & priorities

- **Functional Analysis**
  - AEC selected to provide requirements management
    - RDD-100 Systems Engineering Software
    - Functional hierarchy being developed
    - Requirements traceability & verification
  - "Living" Requirements Document established

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Requirements Definition

Customer

A critical element of the process is the identification of the "customer(s)" and their particular desires for the NTP system. Those customers will consist of the President, Congress, the Nation's taxpayers, NASA management, and other government agencies concerned with the systems development and usage. These customers will most likely have different goals and objectives that must be understood and satisfied. The "voice" of the customers will be required to be part of the requirements definition process to guarantee their requirements are factored into the system.

NTP Requirements

The current top-level requirements for NTP for meeting currently envisioned SEI missions for cargo and piloted Mars missions have been in development over the past two years. A "living" requirements document has been developed with an on-going review process that incorporates current NTP team revisions and suggestions and begins to obtain a complete customer "voice" in the process. The current requirements have been incorporated by Analytical Engineering Corporation (AEC) into Ascent Logic's powerful systems engineering software the Requirements Driven Development (RDD™) System Designer. This will allow for functional analysis, traceability, component-to-functions mapping, model behavior analysis, and failure propagation analysis.

Functional Analysis

AEC will be employing a methodology known as Enhanced Modern Structured Analysis (EMSA) in the analysis of the NTP systems. It will permit a logical structuring of all system functions in a top-down hierarchical decomposition to draw out all the requirements the system must meet while also providing insight for the system-level model developers and technologists. Various options will be provided to display the logical sequences and relationships of operational and support functions that lead to the fulfillment of each NTP function. Time dependent functions will be coupled with behavior models to allow for time-critical functional analysis. This analysis will also develop the basis for establishing functional interfaces and identify system relationships required in meeting SEI mission goals.

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System Definition

- **NTP System Alternatives**
  - Initial database developed on four concepts
  - Date developed based on consistent requirements
- **Reference Concept**
  - Quantify alternative concept benefits
    - Risk - Cost - Schedule - Performance
  - NERVA-derived initial reference due to significant database
- **Evaluation Criteria**
  - Must focus on customer needs
  - Participative approach
  - Utilize Quality Functional Deployment (QFD)
    - TQM tool • Structured & Systematic
    - Expanded usage possible for later program phases

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System Definition

NTP System Alternatives

Efforts were funded in 1992 by NASA to develop consistent state-of-the-art NTP concept data based on the same mission and engine requirements to permit an apples-to-apples comparison. Four alternative concepts were examined by various contractors to evaluate concept feasibility, thrust level implications in the range of 25,000 to 75,000 lbf, test facility requirements, manned mission impacts, key component technologies required, and an industrial approach to developing the system within the next decade. The four concepts examined were each defined based on a specific nuclear fuel element concept consisting of NERVA - derived, CERMET, Particle Bed, and a "twisted-ribbon" fuel element developed by the CIS.

Reference Concept

A reference concept will be utilized to help determine quantitative benefits of alternative engine concepts or subsystem. Significant past efforts on the NERVA concept combined with well understood improvements makes the current NERVA-derived concept the logical choice for the initial reference engine. The use of a reference concept will help in determining the benefits of alternative approaches to better quantify the risk, cost, performance, and schedule impacts.

System Attributes

The process required for evaluation and selection of a single NTP concept must be able to provide a structure that encourages the participation of many various disciplines and provides a focus on the customer needs. The attributes will not be honored if they are not obtained in a participative manner. Quality Functional Deployment, also referredto as the "house of quality," has demonstrated an advantage in providing a systematic and structured approach to achieving high quality systems. QFD identifies the most important system characteristics, relates characteristics directly to requirements, and identifies which characteristics need to be controlled. The current process will concentrate on only providing a system attributes matrix for NTP concept evaluation due to the extensive training, "cultural shock," and laborious nature in implementing QFD. But, with the goal within NASA to provide "faster, better, and cheaper" systems through Total Quality Management (TQM), the initial use of QFD can be expanded to provide the discipline required to achieve this ambitious goal.

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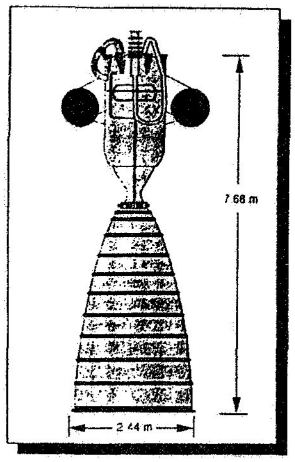
Reference NTP Engine



Characteristics

Flow Cycle	Expander
Fuel Form	Composite (NERVA)
Thrust (lbf)	50,000
Chamber Temperature (K)	2700 (4860 °R)
Chamber Pressure (psia)	785
Nozzle Expansion Ratio	200:1
Specific Impulse (sec)	900
Mass (kg)*	5237
Thrust/Weight	4.3

* Dual Turbopump w/ internal shielding mass (960 kg)

Key Features

- Proven Technology --> Low Risk Approach
- Optimized Engine packaging/nozzle design for maximum Isp
- Optimized flow balance to minimize pressure/weights

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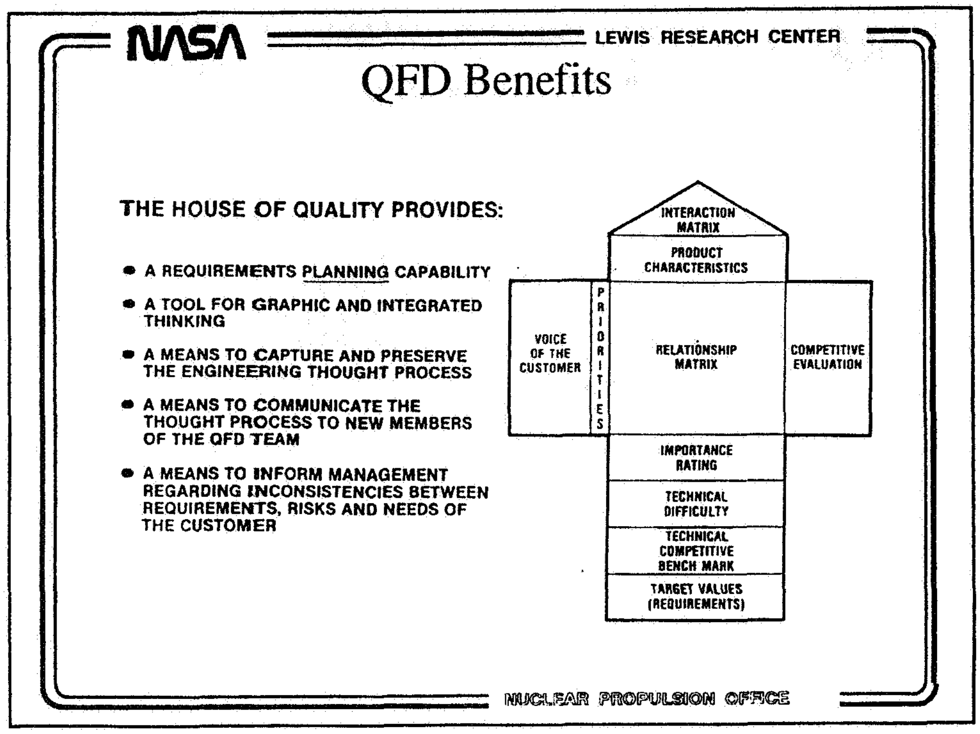
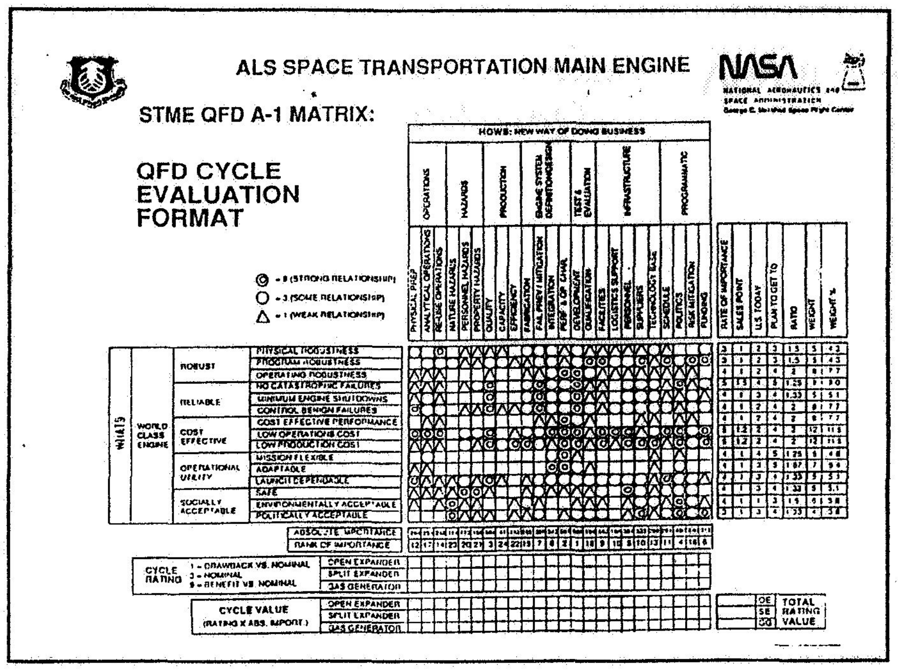
Reference NTP Engine

The reference NTP concept shown above was defined by the Rocketdyne/Westinghouse team. The reference concept is based on a 50,000 pound engine utilizing dual turbopumps, 200:1 nozzle expansion and composite fuel within the NERVA fuel element configuration operating at 2700 K and a 785 psi chamber pressure. This NERVA reference engine shown is preliminary at this point. An initial reference engine and associated database will be determined in the next few months.



QFD Benefits

QFD was developed in Japan in the late 1960's in response to a recognized lack of "quality" in the definition/design process. The foundation for QFD is in the belief that systems should be designed to reflect customer needs and desires, thus requiring all disciplines to work closely together from the time a system is first conceived. Quality Functional Deployment, also referred to as the "house of quality," has demonstrated an advantage in providing a systematic and structured approach to achieving high quality systems. QFD identifies the most important system characteristics, relates characteristics directly to requirements, and identifies which characteristics need to be controlled. QFD provides a significant number of benefits in obtaining a quality product. Some of those benefits are shown above.



QFD Evaluation Matrix Example

The QFD matrix, as shown in the example developed in the space transportation main engine (STME) program, begins with the customer needs, or wants, in phrases that describe the system and its characteristics in their own words. The wants are often grouped into areas of overall customer concerns that typically can include primary, secondary, and tertiary levels. Not all preferences are equal and the customer's needs must be weighted based on discussions with the customers. The top of the QFD matrix lists those engineering characteristics that are likely to affect one or more of the customer needs. These characteristics should describe the system in measurable terms. The body of the matrix is filled with symbols indicating the strength of the customer needs in relationship with the engineering characteristics. On the right-hand side of the matrix, current reference concept's level of meeting customer expectation and opportunities for improvement are determined. The rating of customer needs along with the number and strength of the matrix relationships provides the weighting for the engineering characteristics.

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Consistent Comparison

- **Integrated Government Team Formed**
  - Develop & implement system performance modeling
  - State-of-the-art computational techniques
- **Provide and/or Verify Quantitative Data**
- **Perform Risk and Failure Analysis**
- **Utilize Established Government Cost Models**

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Consistent Comparison

The consistent comparison element of the process must provide and/or verify the quantitative data upon which the concepts will be evaluated. This data must be based on consistent assumptions, groundrules, and requirements. The data provided must also be independently verified to ensure proper analysis has been completed. The fundamental tools that assist the systems engineer in this process are the system performance and cost models, and quantitative risk assessments.

An integrated Government team has been formed to develop and implement a strategy for modeling NTP system performance. The modeling team was formed in order to integrate state-of-the-art computational resources and techniques, along with a diverse knowledge base, into simulations of NTP system performance. A parametric NTP model will be used to predict the system performance for all defined NTP concepts on a consistent basis. The model will also provide steady-state performance data for use in SEI mission analysis and evaluate system design perturbations. Transient evaluations, such as start-up and shut-down, will also be performed as the data and models become available. This will provide a means to evaluate the quantitative benefits to the system based on proposed subsystem and component improvements.

Risk, schedule, and cost analysis will be performed in addition to the performance assessments. The RDD™-100 systems engineering tool will be coupled with the Failure Environment Analysis Tool (FEAT) to assist in the identification of hardware and software failure effects on the entire system. This will ensure that the concept complies to redundancy, reliability, and safety requirements. Cost analysis will utilize established Government cost models to quantify cost benefits to the system upon the implementation of an alternative.

NUCLEAR THERMAL PROPULSION

TECHNOLOGY

N93-26927

NP-TIM-92

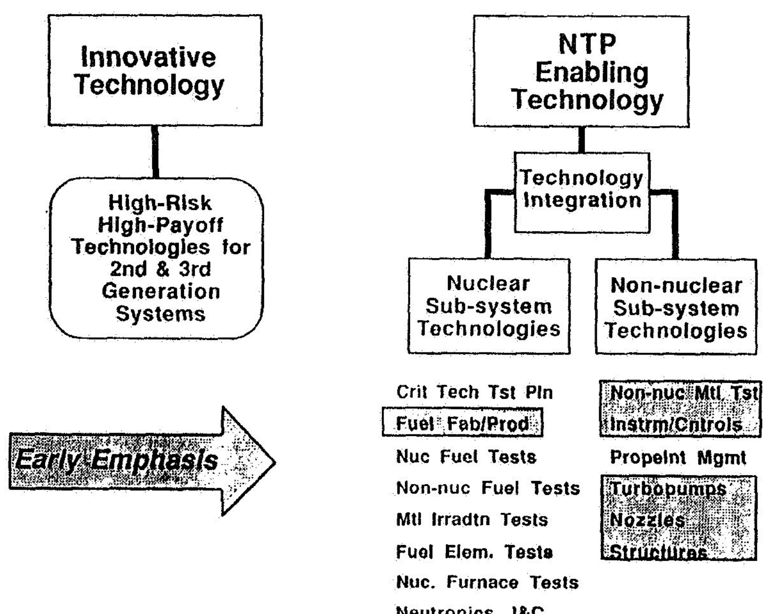
Nuclear Propulsion
Technical Interchange Meeting - 1992
NASA-Lewis Research Center, Plum Brook Station
Sandusky, Ohio, October 20-23, 1992

Nuclear Thermal Propulsion Technology Overview

James R. Stone

NASA/LeRC Nuclear Propulsion Office

NUCLEAR THERMAL PROPULSION TECHNOLOGY ELEMENTS



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NTP Focused Technology Status
Innovative Technology

- Model Development: Graduate Research
- Vapor-Core Modeling & Experiments: INSPI
- Gas-Core Simulation Facility: LeRC
- PBR Stability Modeling: MIT
- PBR Materials Modeling: Univ. New Mexico

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NTP Focused Technology Status
Enabling Technology

- NOZZLES
  - CFD Model Development (3-D Navier-Stokes): LeRC
  - Nozzle Alternatives & Optimization Experiments: LeRC
  - Molecular CFD Plume Model Development: LeRC
- THURBOPUMPS
  - Low-NPSH Pumping Technology: MSFC
  - Materials Evaluation: MSFC
  - 3-D Navier-Stokes CFD Model Development: LeRC
- STRUCTURES
  - Probabilistic Model Development: LeRC
- INSTRUMENTATION & CONTROLS
  - High-Temperature Sensors: LeRC
- MATERIALS
  - Preliminary Sample Prep & Expts: LeRC & MSFC

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Non-nuclear Material

- Goal is usable materials database
- Results needed early to support design and analysis work
- Advanced and commercially available materials to be studied
- Develop required processing and characterization facilities
- Tie-in with Base R & T work

Instrumentation, Controls, and Health Monitoring

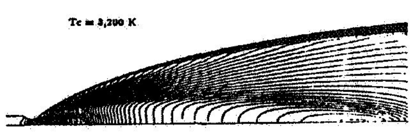
- Large advances since NERVA, needed for autonomous ops
- Details of overall system architecture TBD
- Plan to build off on-going efforts in chemical engine area
- Current LeRC effort concentrating on sensors
- Good progress to date with SiC

Turbopumps

- LeRC working flow and performance modeling
  - Evaluation and modification of existing codes
  - Will use TPA testbed to validate model
- MSFC working hardware specifics
  - Evaluation of concept options, materials, technologies
- Bearing options being studied

Nozzle and Extension

- CFD modeling of internal flow
  - Fluid, thermal, chemical behavior
  - 2-D work done for various temp and thrust ranges
  - Plan to expand results to 3-D, other nozzle forms

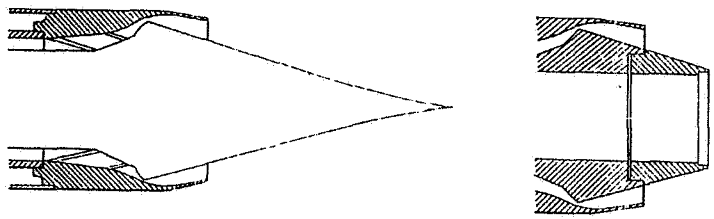


Mach Number Contours of a Reference Nozzle

Static Temperature Contours of a Reference Nozzle

Nozzle and Extension

- Alternative nozzle design evaluation
  - Study of various alternative nozzle forms
  - Goal is performance and packaging improvements
  - Small scale tests to take place in mid 93
  - Promising results flowed back into CFD effort



Nozzle and Extension

- Probalilistic Structural Modeling
  - Large Expansion ratio nozzles (>200:1) cannot be ground tested
  - Develop analytical ability to be able to launch with assured reliability
    - Apply available prob struct modeling methods to NTR nozzle
    - Input CFD results, fabrication process uncertainties
    - Develop probabilistic QA criterion
    - Develop design spec for nozzle and extension

Exhaust Plume Characterization

- Content and behavior of exhaust plume critical to man-ratability
- LeRC developing validated numerical simulation capability
  - CFD not sufficient
  - DSMC with finite difference Boltzmann techniques
- Will accomodate various nozzle shapes, species, conditions
  - Experimental validation planned

Summary

- NTP is key to SEI
- Non-nuclear technologies vital to NTP
- Critical technologies identified
- Work begun, preliminary results available
- Efforts will continue in an evolutionary manner

N93-26928

National Aeronautics and
Space Administration
Lewis Research Center

INSTRUMENTATION & CONTROL
TECHNOLOGY DIVISION

NASA

Silicon Carbide Semiconductor Technology for High Temperature and Radiation Environments

Lawrence G. Matus

Nuclear Propulsion
Technical Interchange Meeting

October 20-23, 1992

Figure 1: This talk will describe silicon carbide technology and its potential for enabling electronic devices to function in high temperature and high radiation environments. The talk will be given by Dr. Lawrence G. Matus of the Instrumentation and Control Technology Division, NASA Lewis Research Center.

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SILICON CARBIDE

A Crystalline material with unique properties

- Abrasive
- Structural
- Refractory
- Semiconducting

Wide energy bandgap
High breakdown voltage
Low dielectric constant
High thermal conductivity
Able to dope both N and P type
High saturated electron drift velocity

CD-90-50908

Figure 2: Silicon carbide (SiC) has many unique properties. The LeRC research program is exploring the semiconductor properties of SiC. The wide energy bandgap of SiC allows it to function at high temperatures, the high breakdown voltage and high thermal conductivity of SiC suggests that power devices and radiation hard devices will be possible, and the low dielectric constant and high saturated electron drift velocity of SiC opens the possibility of high frequency devices. SiC can be doped both n- and p-type for electronic device fabrication.

COMPARISON OF SEMICONDUCTORS

Property	Si	GaAs	GaP	3C SiC (6H SiC)	Diamond
Bandgap (eV) at 300 K	1.1	1.4	2.3	2.2 (2.9)	5.5
Maximum operating temperature (K)	600	760	1250	1200 (1580)	1400 (?)
Melting point (K)	1690	1510	1740	Sublimes >2100	Phase Change
Physical stability	Good	Fair	Fair	Excellent	Very Good
Electron mobility R.T., $cm^2/V\text{-}s$	1400	8500	350	1000 (600)	2200
Hole mobility R.T., $cm^2/V\text{-}s$	600	400	100	40	1600
Breakdown voltage E_b, 10^6 V/cm	[illegible]	.4	[illegible]	4	10
Thermal conductivity σ_T, W/cm-°C	1.5	.5	.8	5	20
Sat. electron drift vel. v(sat), 10^7 cm/s	[illegible]	2	[illegible]	2.5	2.7
Dielectric const., K	11.8	12.8	11.1	9.7	5.5

CD-91-55350

Figure 3: The properties of the two most common polytypes of SiC (3C-SiC, also called beta or cubic SiC and 6H-SiC, also called alpha SiC) are compared with the properties of commercially available semiconductors and diamond. The commercially available semiconductors were judged unable to meet the 600°C temperature goal the LeRC program, while diamond technology was considered to be too far in the future.

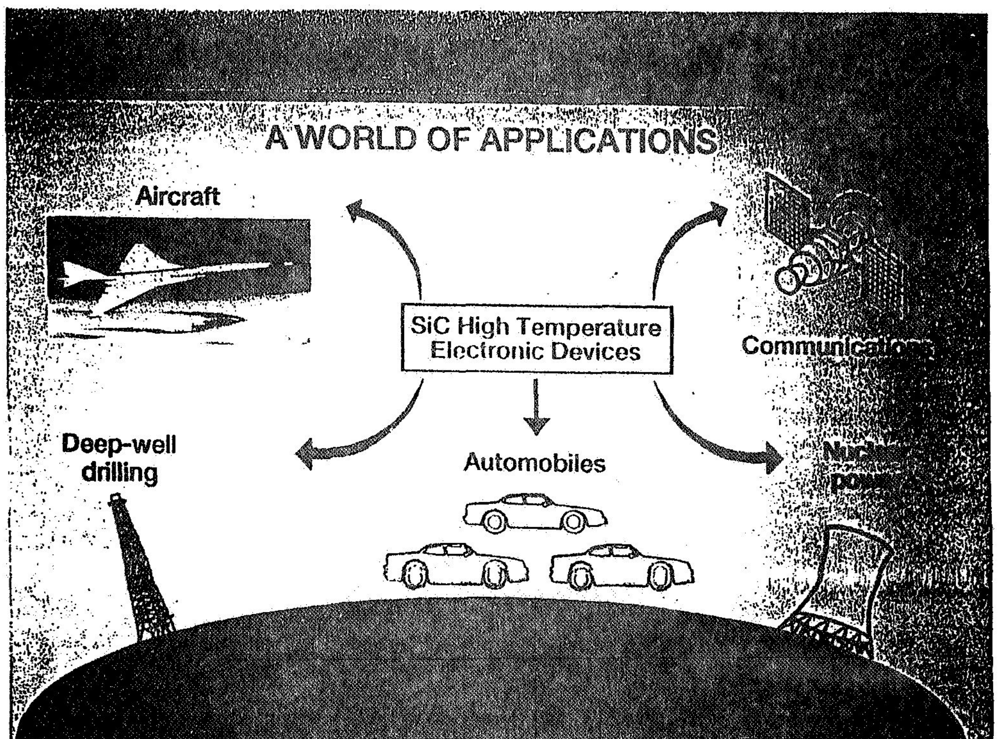

Figure 4: High temperature and/or radiation hard electronic devices fabricated from SiC would have a world of aerospace and terrestrial based applications.

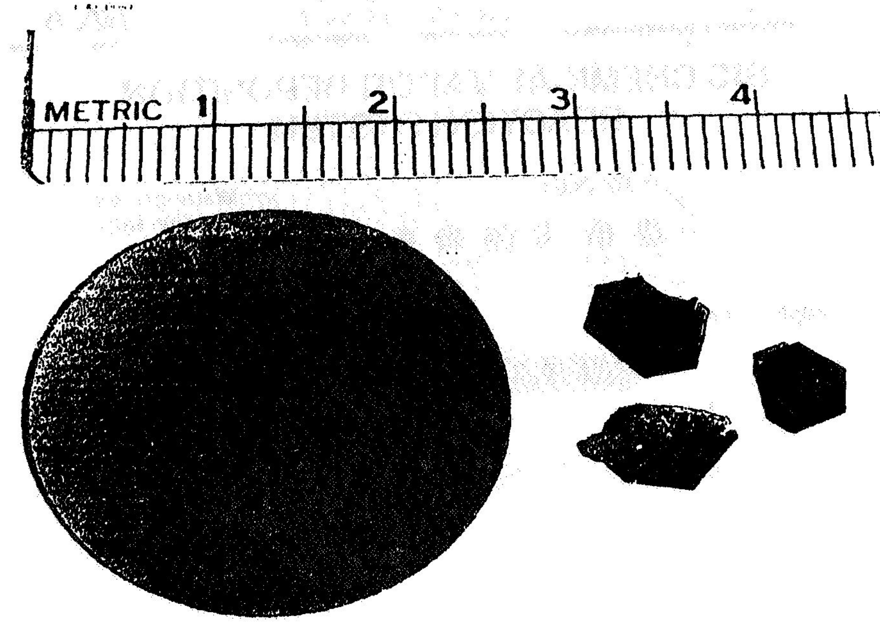

Figure 5: Prior to 1989, SiC researchers had to use small irregular-shaped Lely SiC samples for device studies. Now, Cree Research, Inc., a small company in Durham, North Carolina, has made one inch SiC substrates commercially available.

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SUBLIMATION GROWTH OF 6H-SiC BOULES

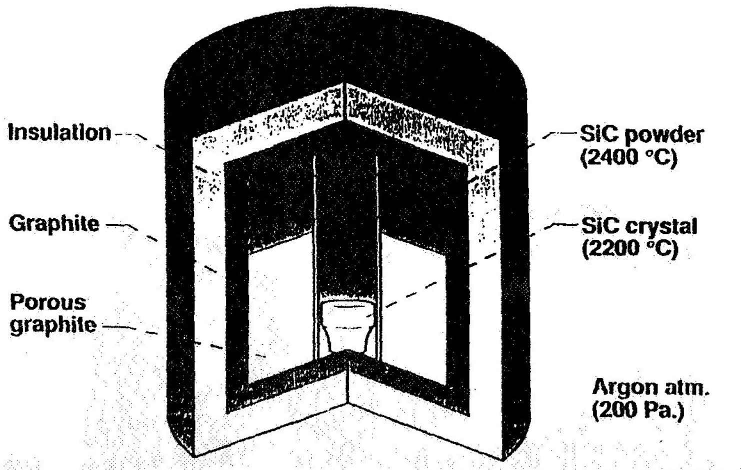


Figure 6: The difficulty in producing SiC substrates is that SiC does not melt. Therefore, the SiC boule crystal growing technique involves the sublimation of SiC powder. The SiC boule growth is carried out in a high temperature furnace where after the SiC powder sublimes, the SiC vapor is transported along a temperature gradient, and then deposits onto a SiC seed crystal which is at a cooler temperature.

SiC CHEMICAL VAPOR DEPOSITION REACTION SYSTEM

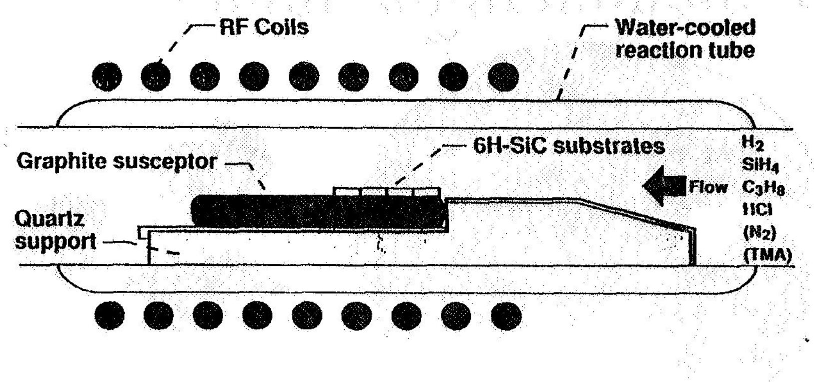


Figure 7: During the 1980s, the LeRC SiC program developed a chemical vapor deposition technique for the heteroepitaxial growth of 3C-SiC onto silicon substrates. This 3C-SiC material has many defects because of the mismatch in material properties between the SiC and silicon. However, the chemical vapor deposition process works well for the homoepitaxial growth of 6H-SiC onto 6H-SiC substrates. Our process uses silane as the silicon source, propane as the carbon source, and hydrogen as the carrier gas. For doping the SiC epilayers, nitrogen gas produces n-type while trimethylaluminum vapor produces p-type SiC. Hydrogen chloride gas is used during a pregrowth etch. The growth temperature is 1450°C. A radio-frequency generator heats the graphite susceptor. The growth process takes place at atmospheric pressure.

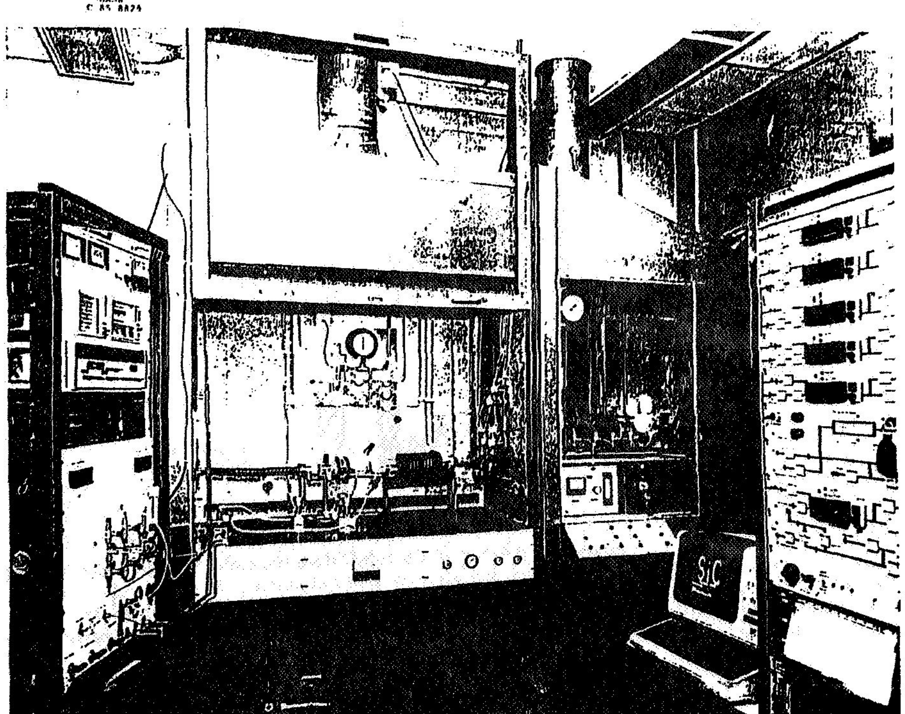
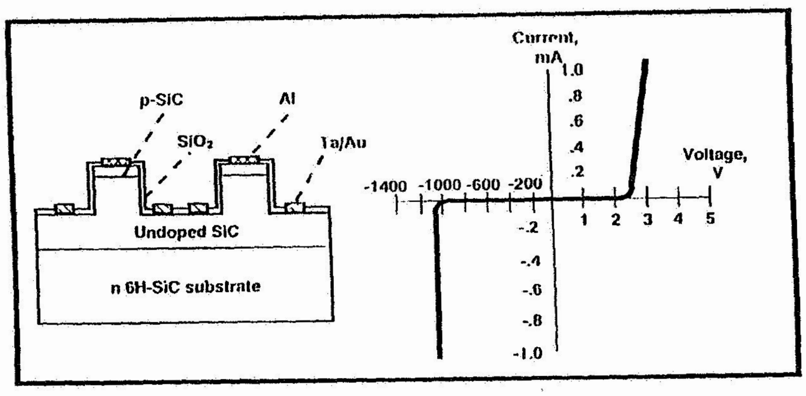


Figure 8: A photo of the LeRC chemical vapor deposition system. The time sequence and flow rates of all process gases are computer controlled.

6H SILICON CARBIDE P-N JUNCTION DIODE



Diode Structure

I-V Characteristics at Room Temperature

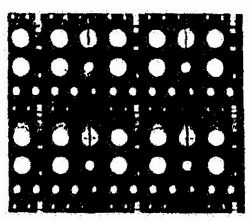
CD-91-55351

Figure 9: Device structure and room temperature current-voltage characteristics for a 6H-SiC pn junction diode that was fabricated at LeRC. Etching of the SiC to produce the mesa-style pn junction configuration was accomplished by reactive ion etching; there are no known wet chemical etchants for SiC. The reverse diode characteristics demonstrated very low leakage out to the breakdown voltage of around 1000 volts. The forward diode characteristics displayed the rather large turn-on voltage associated with the wide bandgap of SiC.

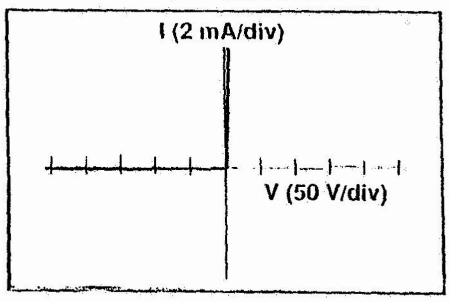
Silicon Carbide Junction Diode

Accomplishment: Highest reported operational temperature (600 °C) for any p-n junction diode device. Significantly improved characteristics above 400 °C. Demonstrates high quality 6H-SiC epitaxial film growth processes.

Diode array



I-V Characteristics at 600 °C



Benefits: Silicon carbide diodes (p-n junctions) are basic building blocks from which all future silicon carbide electronic devices will be developed

CD-91-53902

Figure 10: The diode array photo shows SiC diodes of different sizes. The diode sizes range from 50 to 400 microns in diameter. The 6H-SiC diode demonstrated excellent current-voltage characteristics when tested to 600°C. As a semiconductor material, SiC can clearly perform at elevated temperatures. The SiC pn junction is a basic building block from which all future SiC electronic devices will be developed.

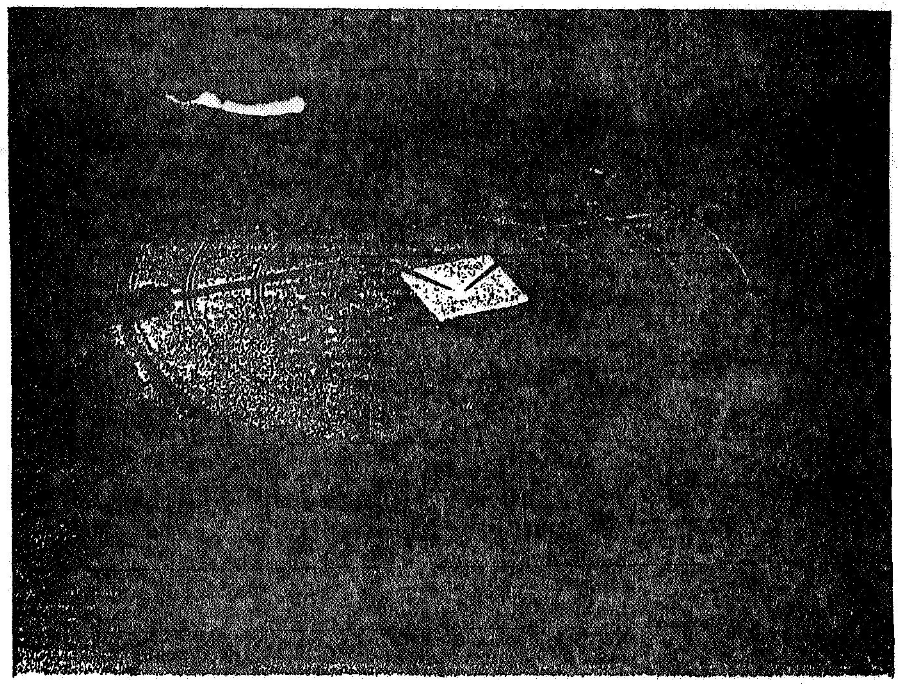


Figure 11: A photo documenting the operation of a 6H-SiC pn junction diode at 600°C. One diode, of the many on the chip, is being examined on a probing station equipped with a heating stage. The forward biased diode is emitting blue light while the heating stage is glowing cherry-red at 600°C.

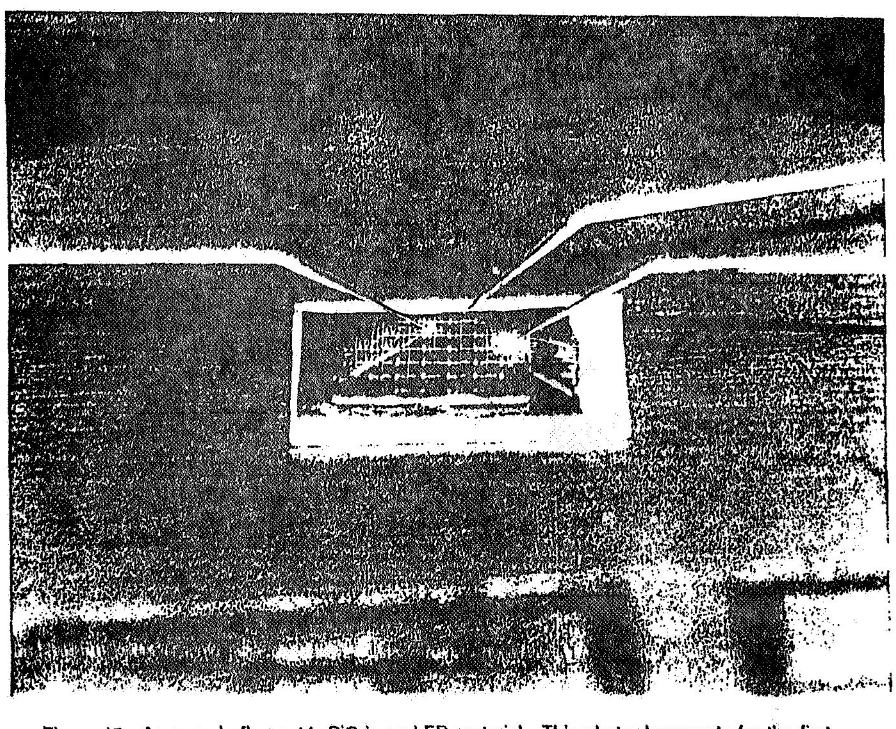


Figure 12: As seen in figure 11, SiC is an LED material. This photo documents for the first time, that both 6H-SiC blue LEDs and 3C-SiC green-yellow LEDs can be produced on a single chip. The ability to fabricate a 3C-SiC LED is an indication of the high quality of the 3C-SiC material.

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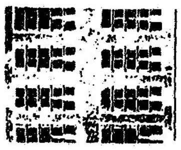
INSTRUMENTATION & CONTROL TECHNOLOGY DIVISION

NASA

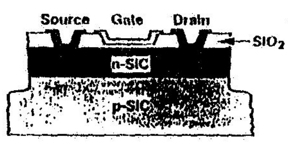
SILICON CARBIDE MOSFET

Milestone: Develop and demonstrate a high temperature, (400 °C), 6H-SiC metal-oxide-semiconductor field effect transistor (MOSFET)

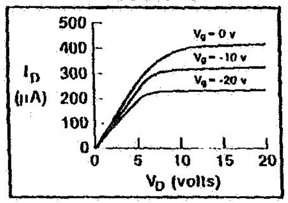
MOSFET Array



SiC MOSFET Structure



I-V Characteristics at 500 °C



Accomplishments: A depletion-mode silicon carbide MOSFET has been developed and successfully demonstrated at an operational temperature of 500 °C.

Benefits: Silicon carbide MOSFETs (switches) provide the most basic active electronic device from which integrated circuits can be developed.

CD-91-55354

Figure 13: A depletion-mode 6H-SiC Metal-Oxide-Semiconductor Field-Effect Transistor (MOSFET) was demonstrated to an operating temperature of 500°C. SiC has silicon dioxide as its native oxide, so many of the silicon oxidation techniques are directly importable to the SiC technology. The current-voltage characteristics for this MOSFET are not yet ideal because the device structure and oxide growth processes have not yet been optimized.

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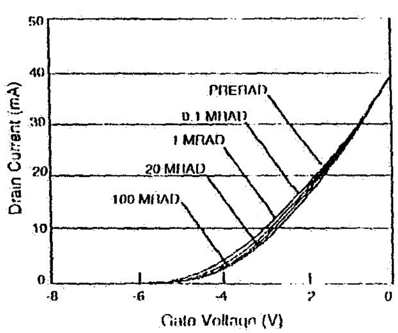
INSTRUMENTATION & CONTROL TECHNOLOGY DIVISION

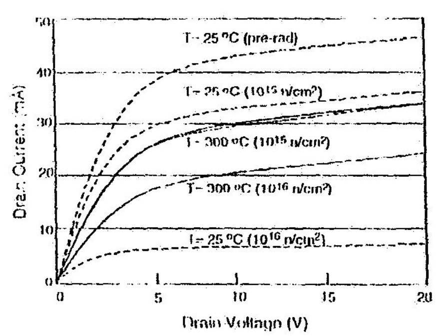
NASA

Silicon Carbide JFET Radiation Response

Gamma



Neutron



Work performed by J.M.McGarrity *et al.* Harry Diamond Laboratories

To be published in IEEE Trans. on Nuclear Science, 39, No.6 Dec. 1992

Figure 14: SiC is expected to be a radiation-hard semiconductor. Work performed at the Harry Diamond Laboratories demonstrates that, yes indeed, SiC is radiation-hard. 6H-SiC Junction Field-Effect Transistors (JFETs) were exposed to both gamma and neutron radiation. The JFET experienced little effect from the gamma radiation and was still functioning after an exposure of 10^{16} neutrons per cm². The JFET also performed better at the elevated temperature of 300°C than at room temperature after the neutron exposure.

AREAS REQUIRING TECHNOLOGY ADVANCEMENT
(FOR HIGH TEMPERATURE APPLICATIONS)

- Metallization (electrical) contacts
- Passivation and dielectric layers
- Wire attachment
- Packaging
- Circuit board technology

Figure 15: Several areas still require technology advancement before SiC is ready for high temperature and/or high radiation environments. The LeRC program is supporting research in the areas of metallization, passivation and dielectric layers, wire attachment, and component packaging. Ultimately circuit board technology must be developed.

NASA
C-90-12561

CONCLUDING REMARKS

- Need for 600 °C electronic devices
- SiC is the semiconductor of choice
- Significant SiC crystal growth progress
- Discrete devices (diodes and MOSFETs) demonstrated
- Several challenging areas await
- SiC is on its way

Figure 16: Concluding Remarks: The LeRC program believes that a need for high temperature (600°C) and/or high radiation-hard electronic devices exists. The semiconductor of choice is SiC because of its many unique properties and the fact that diamond is still far in the future. During the past ten years, significant progress has been made in the advancement of SiC technology. Key progress has been made in the SiC crystal growth process. This progress has allowed device scientists to fabricate prototype SiC electronic devices with exciting characteristics and thus, LeRC researchers feel that SiC, as an electronic material, is definitely on its way. However, as is probably evident, there are still a number of challenging areas of research to be pursued.

N93-26929

National Aeronautics and
Space Administration

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NASA

NTR PLUME MODELING

Presented to the
Nuclear Propulsion Technical Interchange Meeting

October 21, 1992

D. BYERS, Chief, Low Thrust Propulsion Branch

C.-H. CHUNG, Principal Investigator

R. STUBBS, Chief, Computational Methods for Space Branch

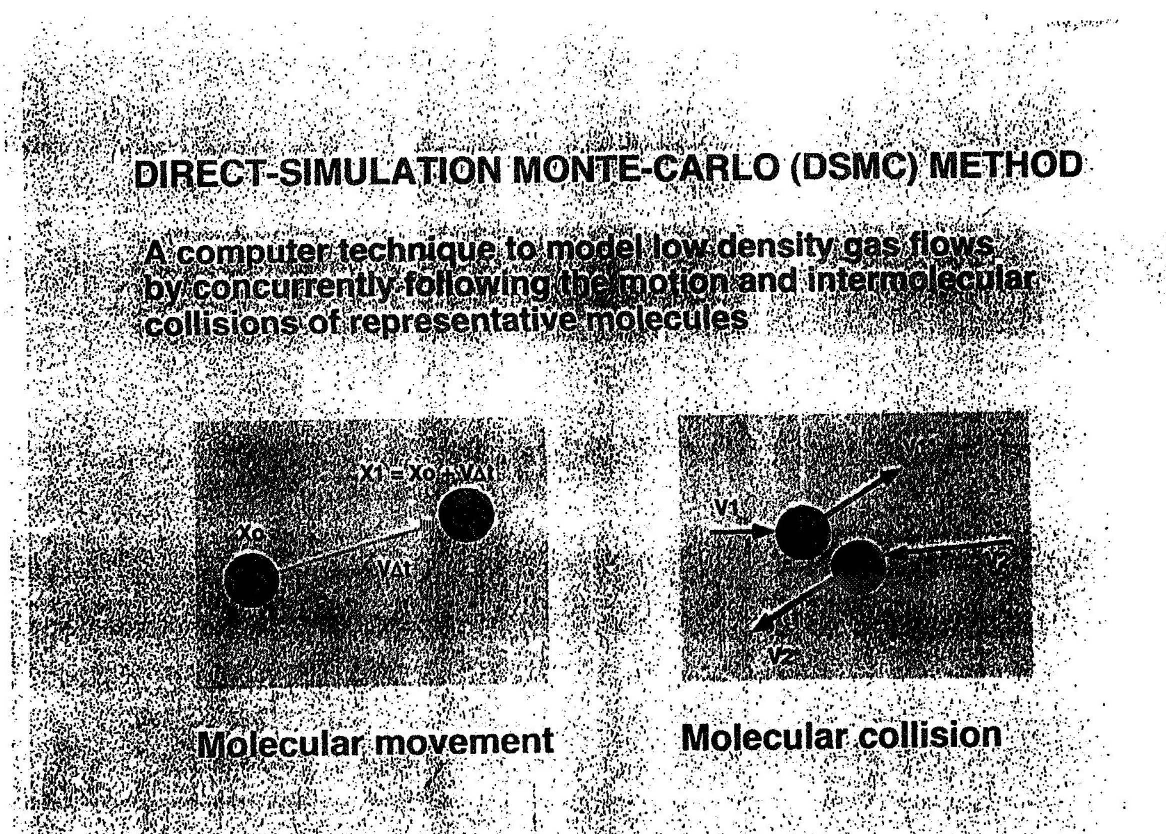
COMPUTATIONAL FLUID DYNAMICS (CFD)
FOR
PLUME ANALYSIS

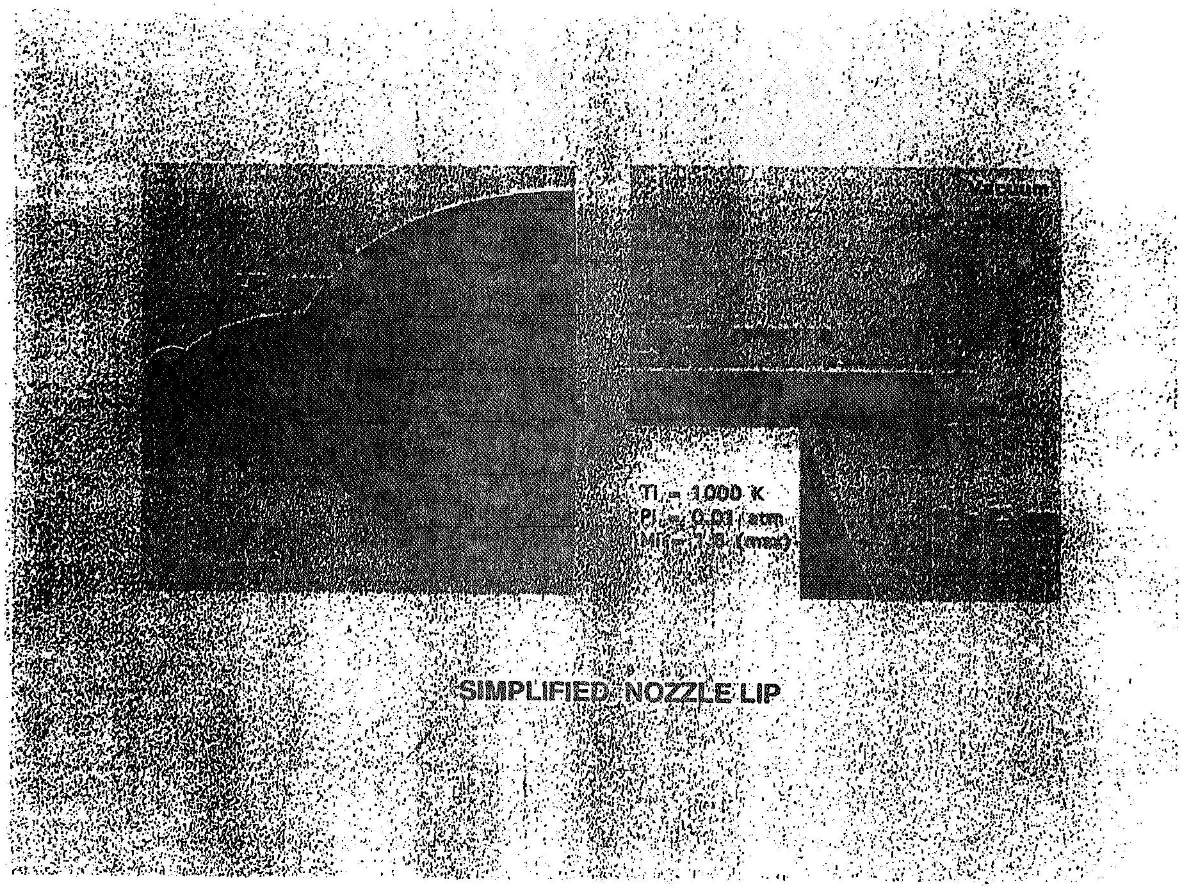
MOLECULAR FLUID MECHANICS

- THE VAST MAJORITY OF CFD DEALS WITH GASES WHICH ARE ADEQUATELY DESCRIBED BY THE CONTINUUM THEORY, I.E., THE NAVIER-STOKES EQUATIONS.
- IN RAREFIED GAS FLOWS, A MOLECULAR MODEL IS APPROPRIATE, REQUIRING DIFFERENT TECHNIQUES.
  - DIRECT SIMULATION MONTE-CARLO (DSMC)
  - FINITE DIFFERENCING OF THE BOLTZMANN EQUATION
- MOLECULAR CFD IS REQUIRED FOR:
  - NOZZLE LIP AND CRITICAL BACKFLOW REGIONS
  - PLUME / SPACECRAFT INTERACTIONS
  - GROUND TESTING

MOLECULAR CFD CHARACTERISTICS

- DSMC TECHNIQUES TRACK A LARGE NUMBER OF MOLECULES (OF ORDER 10^5 TO 10^7) AND MODEL THEIR INTERACTIONS STATISTICALLY.
- COMPUTATIONALLY INTENSIVE
- DR. CHAN-HONG CHUNG HAS DEVELOPED AN ENHANCED DSMC CODE WITH MULTI-SPECIES CAPABILITY, ALLOWING MORE ACCURATE CALCULATIONS OF SPECIE SEPARATION.





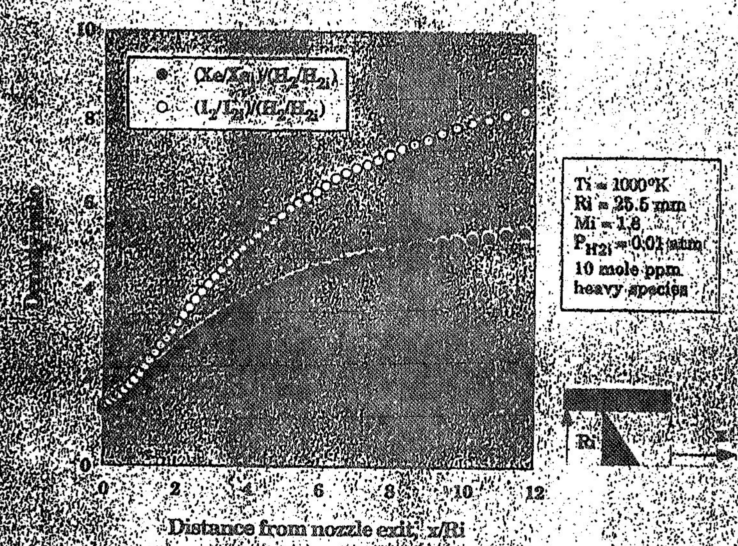
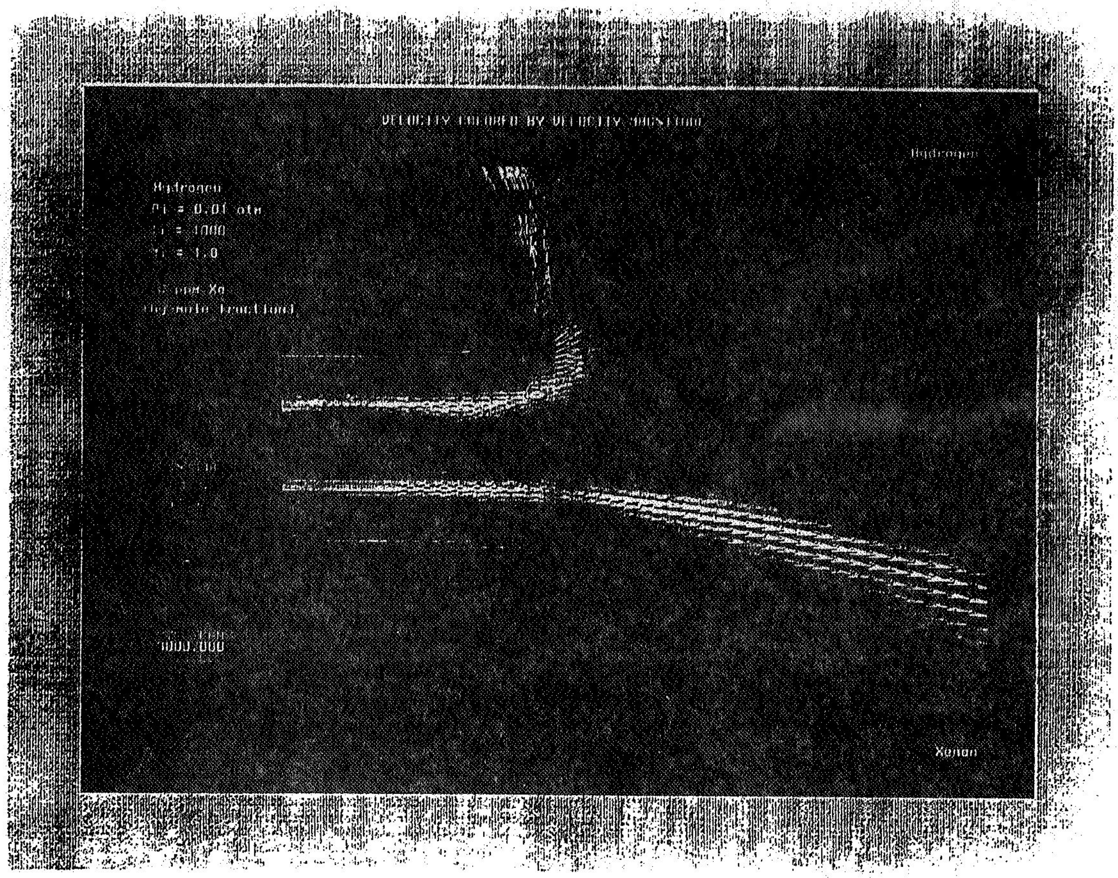


Fig. 2 Effect of molecular weight on species seperation along bottom plane



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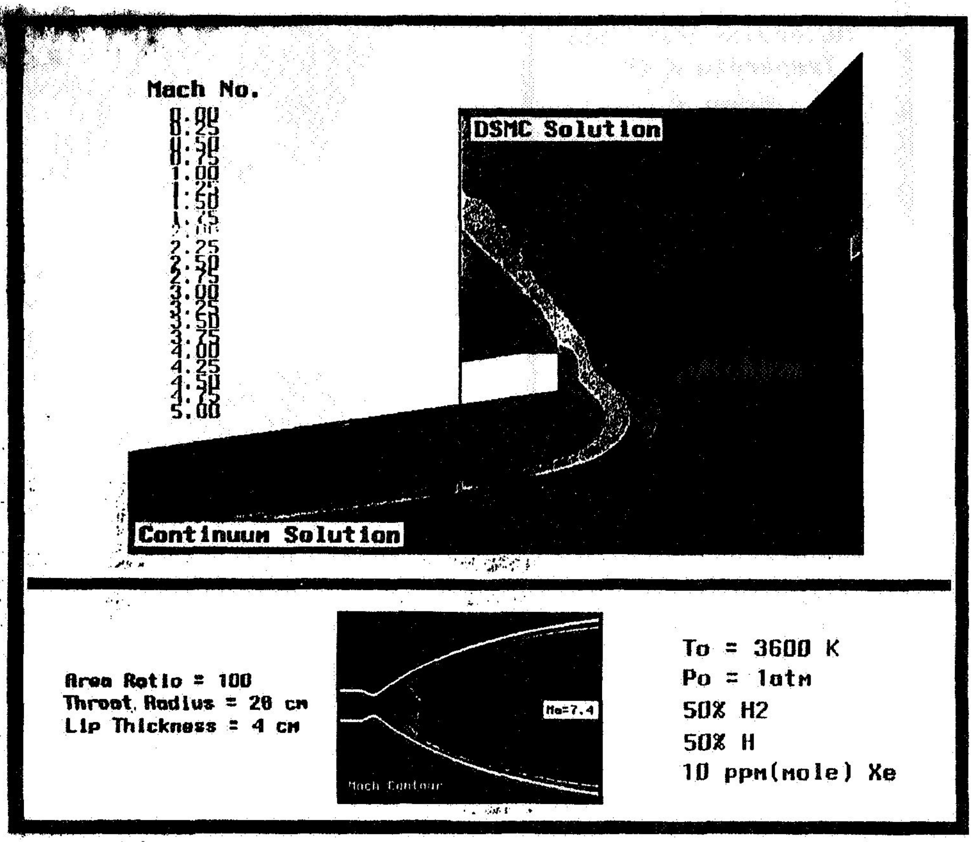
NTR PLUME MODELING

INTEGRATION OF

DSMC AND NAVIER-STOKES COMPUTATIONS



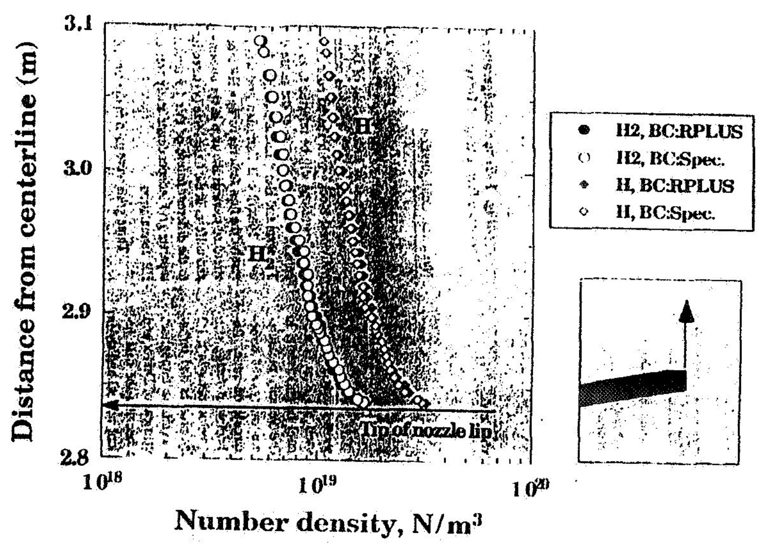
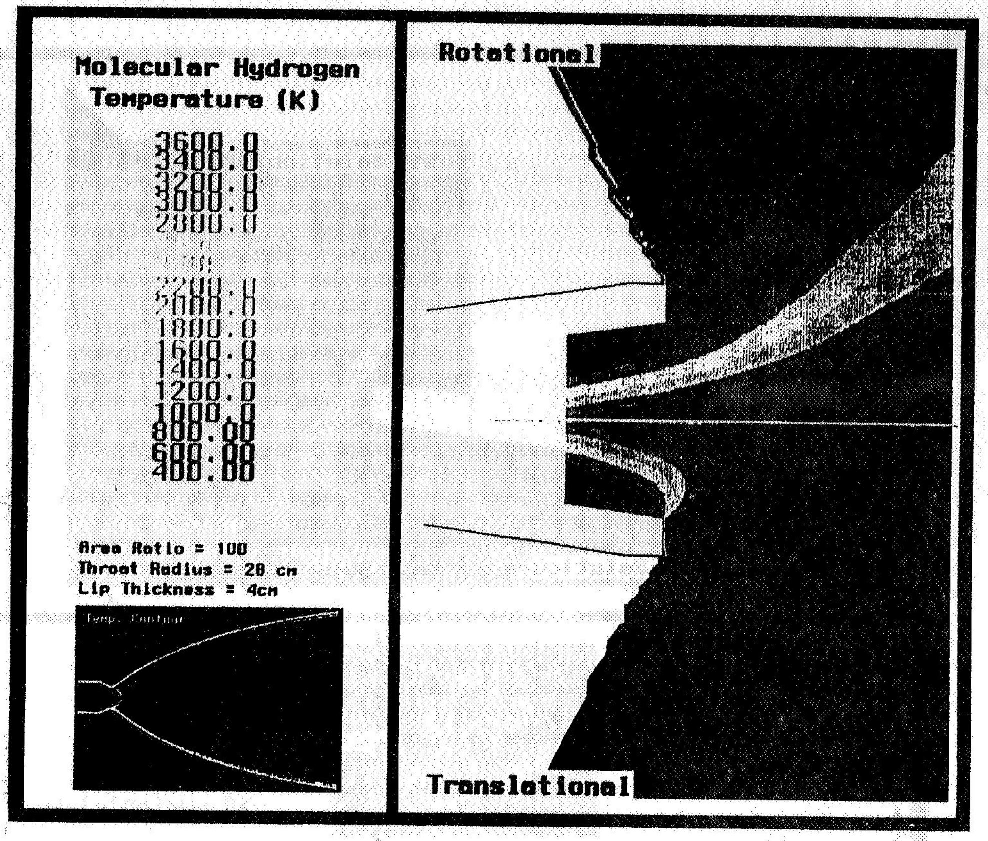


Fig.1 Density profile along the line parallel to exit plane



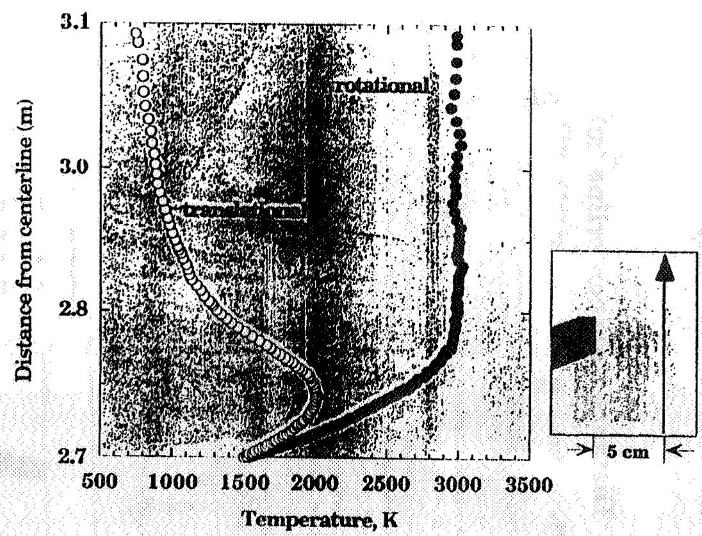
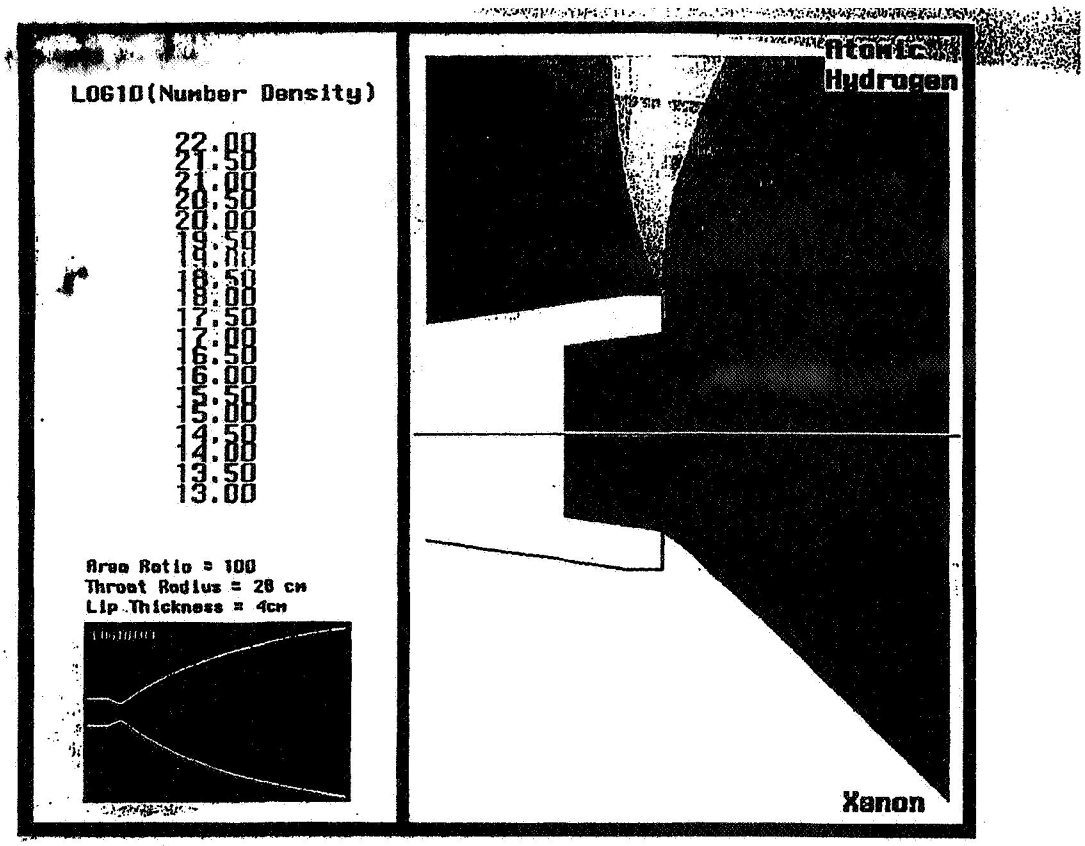
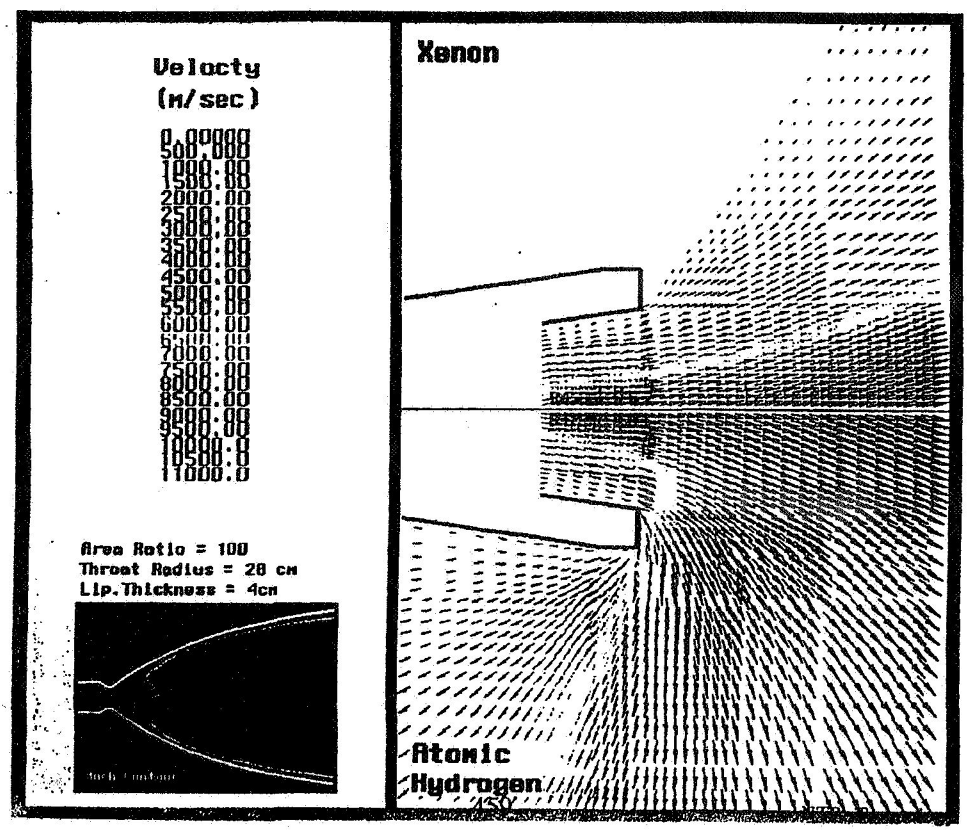


Fig.4 Molecular hydrogen temperature profile along the line parallel to exit plane

C-6





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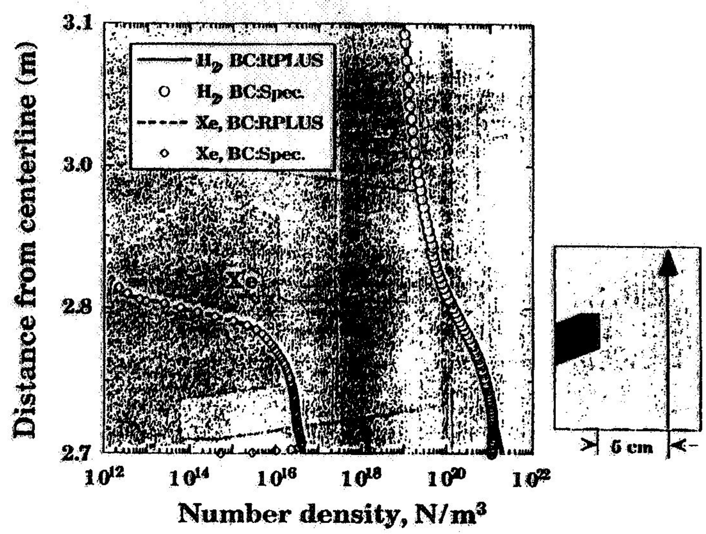

Fig.2 Density profile along the line parallel to exit plane

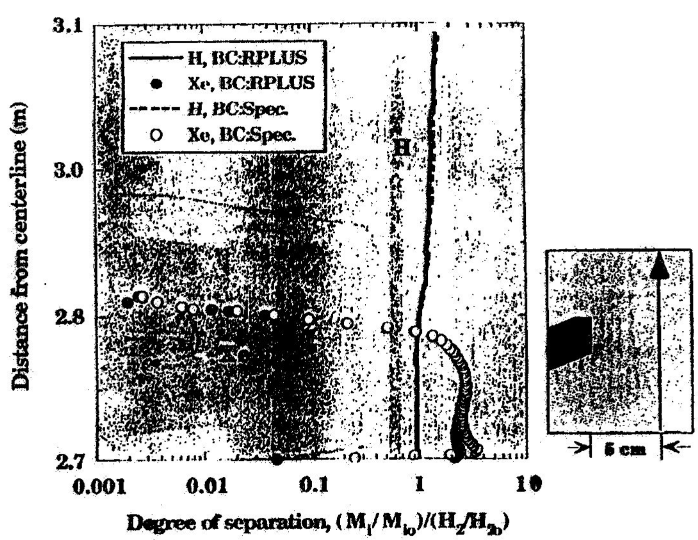

Fig.3 Degree of separation along the line parallel to exit plane

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N93-26930

National Aeronautics and
Space Administration

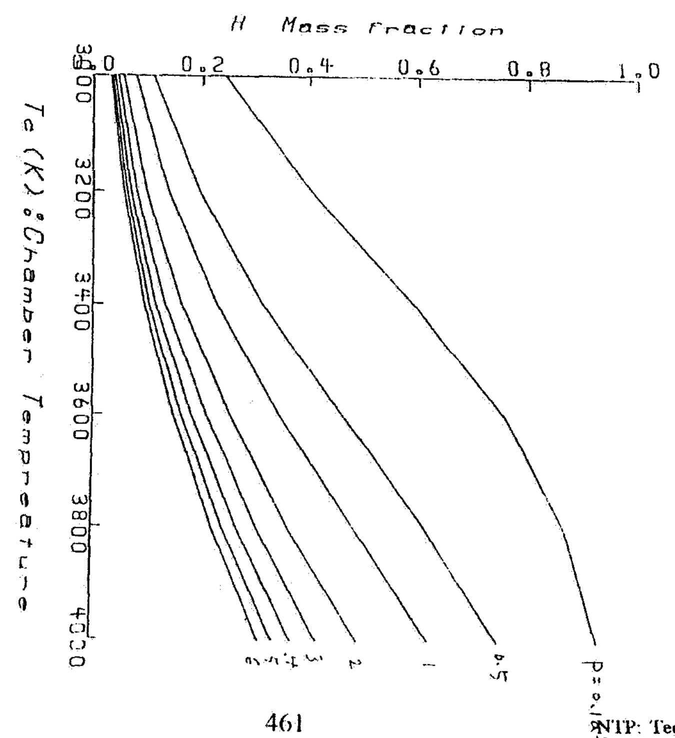
Lewis Research Center

COMPUTATIONAL FLUID DYNAMICS FOR NUCLEAR THERMAL PROPULSION

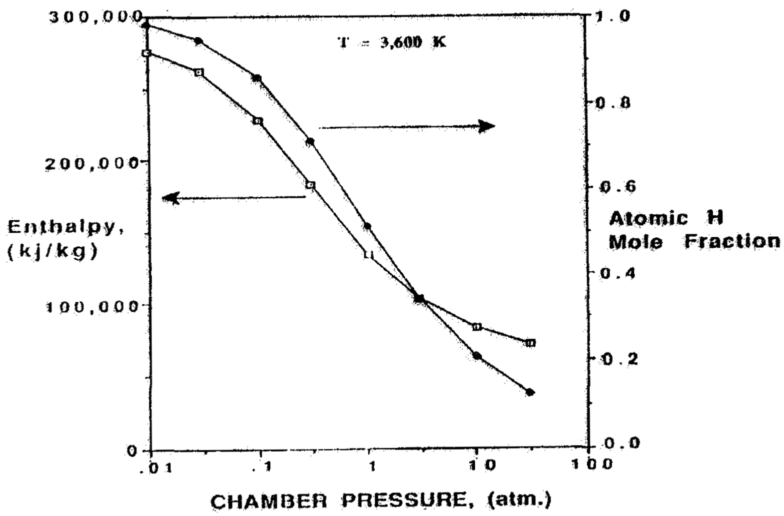
Presented to the
Nuclear Propulsion Technical Interchange Meeting

October 21, 1992

Robert M. Stubbs
Suk C. Kim

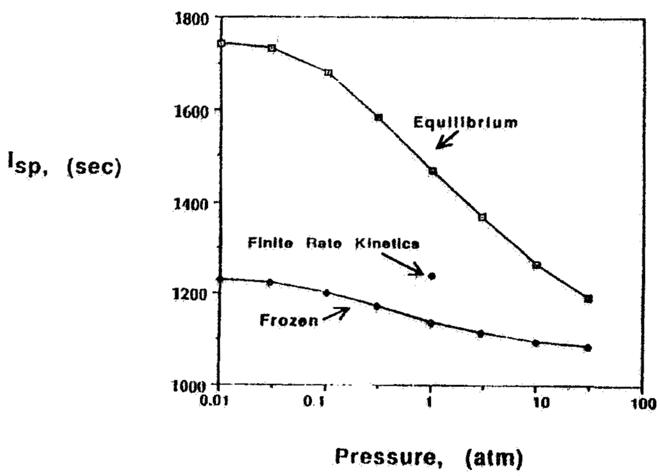


SPECIFIC ENTHALPY OF HYDROGEN AND MOLE FRACTION OF H
AS A FUNCTION OF CHAMBER PRESSURE
AT A CHAMBER TEMPERATURE OF 3,600 K



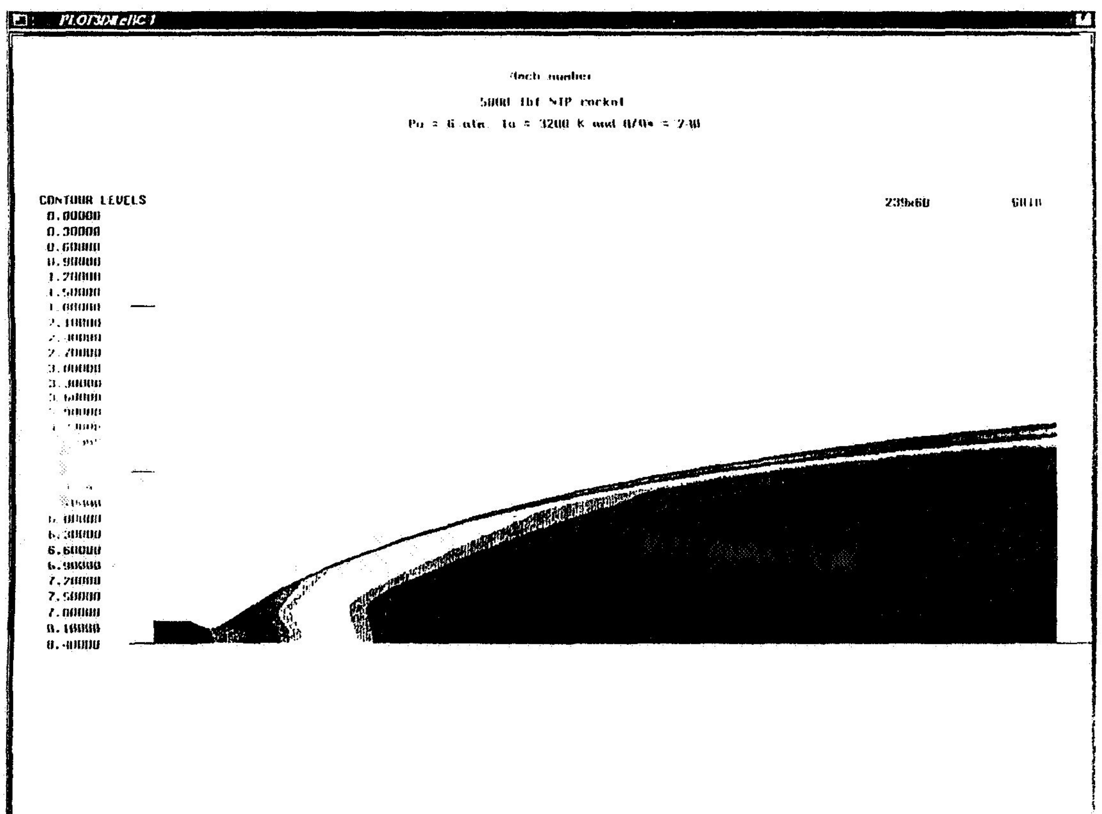
SPECIFIC IMPULSE AS A FUNCTION OF CHAMBER PRESSURE

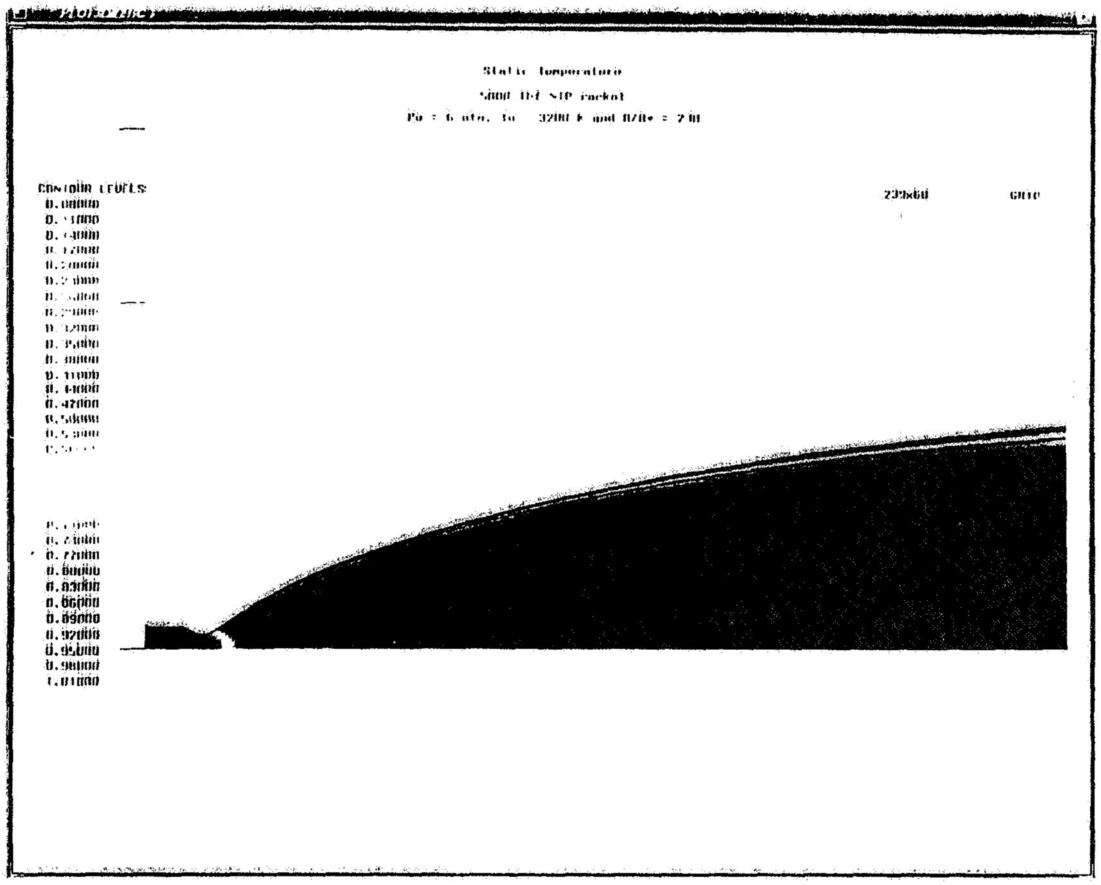
T_C = 3,600 K

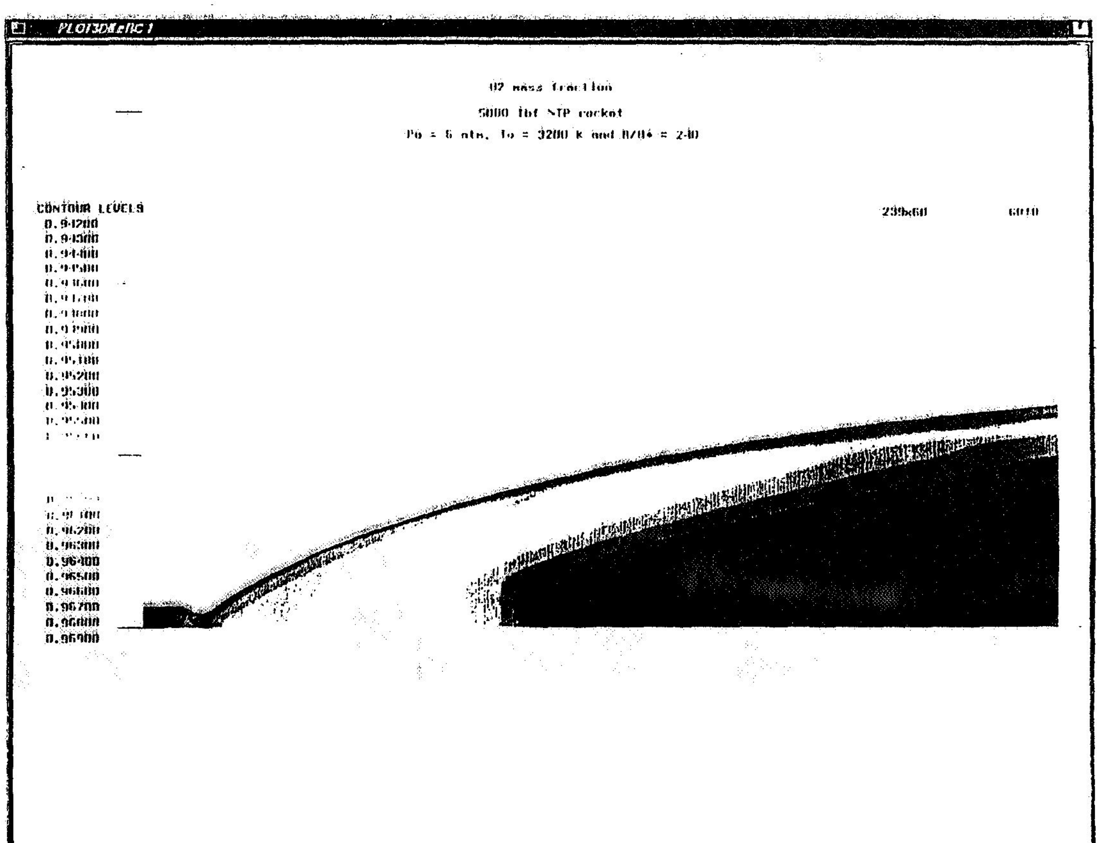


RPLUS

- DEVELOPED AT NASA-LEWIS
- A NAVIER-STOKES CODE WITH FINITE RATE CHEMICAL KINETICS CAPABILITY
    - LU-SSOR
    - 9 SPECIES, 18 REACTIONS, (H_2, O_2 COMBUSTION SYSTEM)
    - 3-D, (ONLY 2-D AXISYMMETRIC REQUIRED HERE)

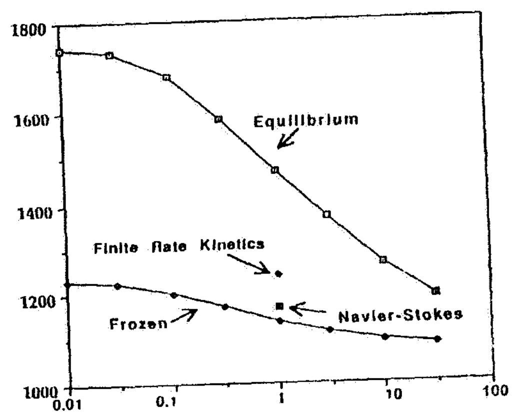



Static Temperature
5000 lbf NTP rocket
CONTOUR LEVELS
239x60
GRID


PLOT3D/FAST 1
mass fraction
5000 lbf NTP rocket
CONTOUR LEVELS
239x60
GRID

SPECIFIC IMPULSE AS A FUNCTION OF CHAMBER PRESSURE

T_C = 3,600 K

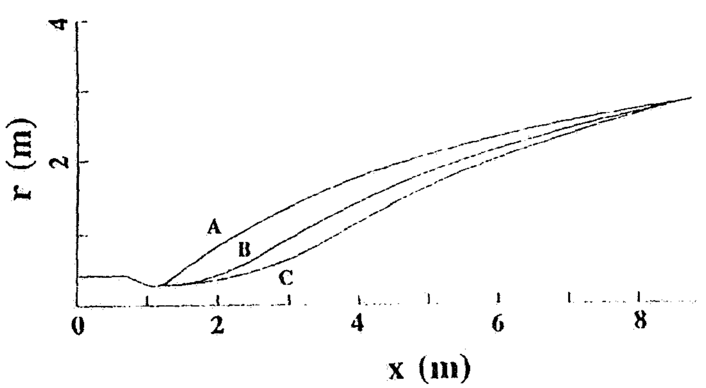


NASA LEWIS RESEARCH CENTER

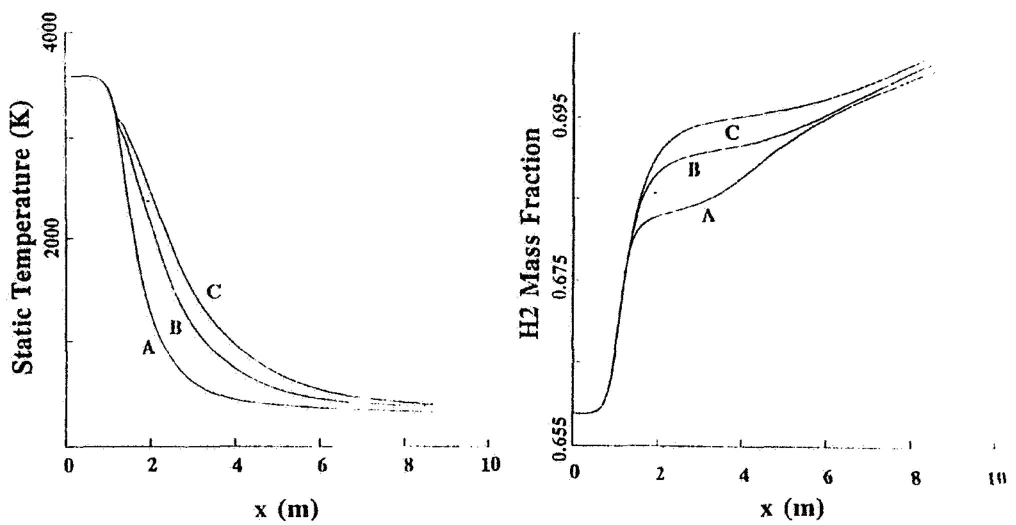
WALL CONFIGURATIONS OF NOZZLES "A", "B", AND "C"

ALL HAVE:

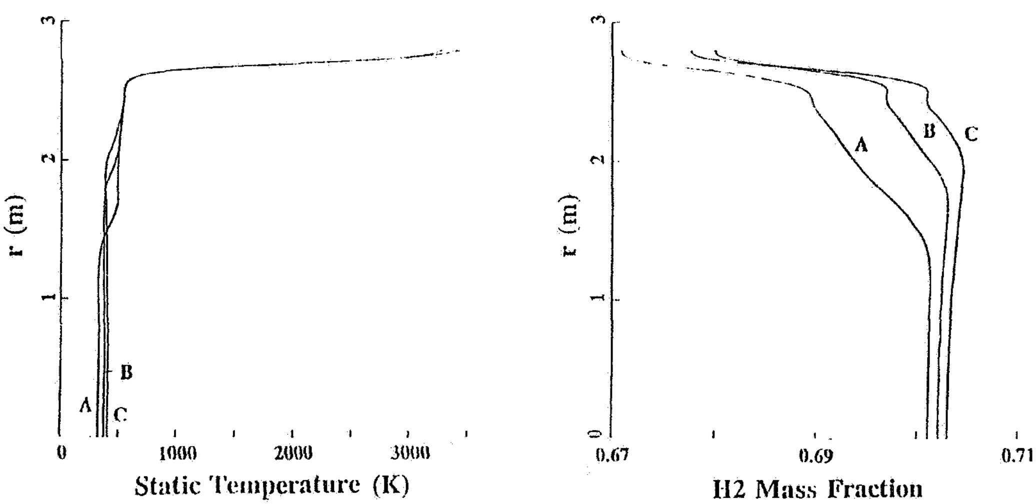
- R_{THROAT} = 0.28 m
- A_E/A_T = 100
- THROAT TO EXIT LENGTH = 7.6 m

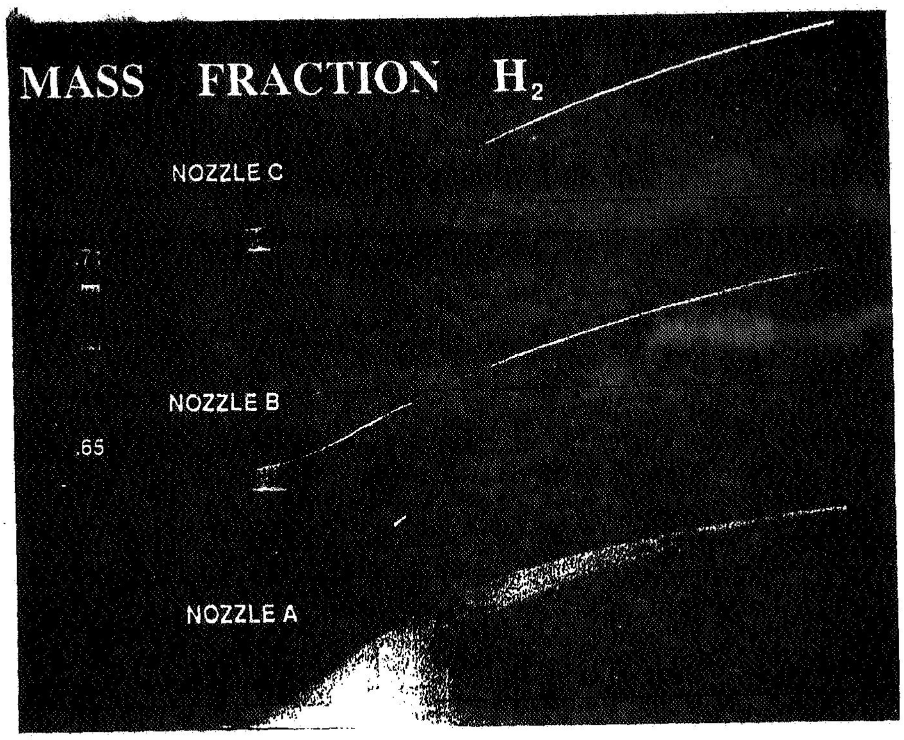


Axial Distributions on the Centerlines



Radial Distributions at the Exit





National Aeronautics and
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TABLE 4. Specific Impulse of NTP Nozzles which have been scaled to produce, at each Temperature, approximately equal Thrust.

	Isp, (lb_f-s/lb_m)		
	P_c=10 atm	P_c=1.0 atm	P_c=0.1 atm
T_c, (K)	r_t =0.28 m	r_t=0.8854 m	r_t=2.8 m
2700	901.61	899.48	903.14
3200	1024.33	1037.21	1072.47
3600	1144.22	1183.39	1223.17

TABLE 5. Specific Impulse for variously sized NTP Nozzles with T_c=3600 K, P_c=1.0 atm.

Isp, (lb_f-s/lb_m)		
r_t =0.28 m	r_t=0.8854	r_t=2.8 m
1151.57	1183.39	1220.41

COMPUTATIONAL FLUID DYNAMICS
FOR
NUCLEAR THERMAL PROPULSION

SUMMARY

- CFD SIMULATIONS PREDICT LOWER SPECIFIC IMPULSE VALUES FOR THE LOW PRESSURE NUCLEAR THERMAL ROCKET THAN ONE-DIMENSIONAL, INVISCID ANALYSES.

- THE LOW PRESSURE CONCEPT SHOWS MORE PROMISE AT HIGHER TEMPERATURES THAN AT LOWER TEMPERATURES, BECAUSE OF THE GREATER AMOUNT OF DISSOCIATION.

- SMALLER NOZZLES SHOW LARGER VISCOUS LOSSES, ESPECIALLY AT LOW PRESSURES; THEREFORE, PERFORMANCE GAINS ARE ASSOCIATED WITH LARGER NOZZLES.

- ADVANCED CFD CODES SUCH AS RPLUS (3D, NAVIER-STOKES, CHEMICAL KINETICS), WITH THEIR ABILITY TO SIMULATE REAL GAS EFFECTS, PROVIDE THE DESIGNER WITH POWERFUL TOOLS TO ANALYZE THE ENTIRE FLOW FIELD AND CALCULATE GLOBAL PERFORMANCE VALUES.

ON-GOING WORK

NOZZLE WALL FILM-COOLING STUDIES

- EFFECTIVENESS OF HYDROGEN FILM COOLING
  - AMOUNT OF HYDROGEN
  - OPTIMIZATION OF FILM PLACEMENT
- PERFORMANCE IMPACT
  - INTERACTION WITH PRIMARY FLOW
  - LOSSES IN SPECIFIC IMPULSE

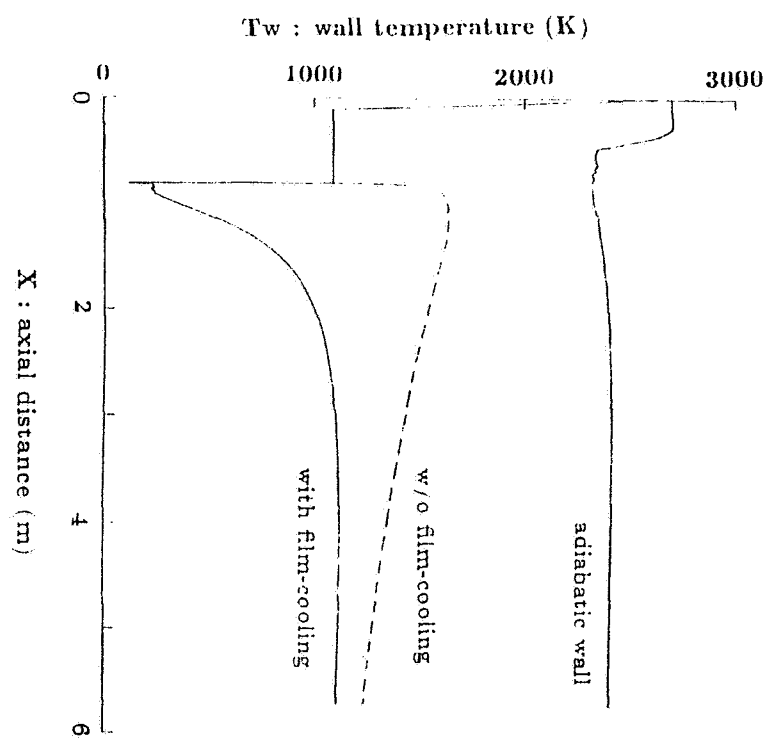


Fig. 4 Temperature variations along the nozzle wall with and without film-cooling.

N93-26931

NP-TIM-92

Nuclear Propulsion
Technical Interchange Meeting - 1992
NASA Lewis Research Center, Plum Brook Station
Sandusky, Ohio, October 20-23, 1992

Probabilistic Structural Analysis for Nuclear Thermal Propulsion

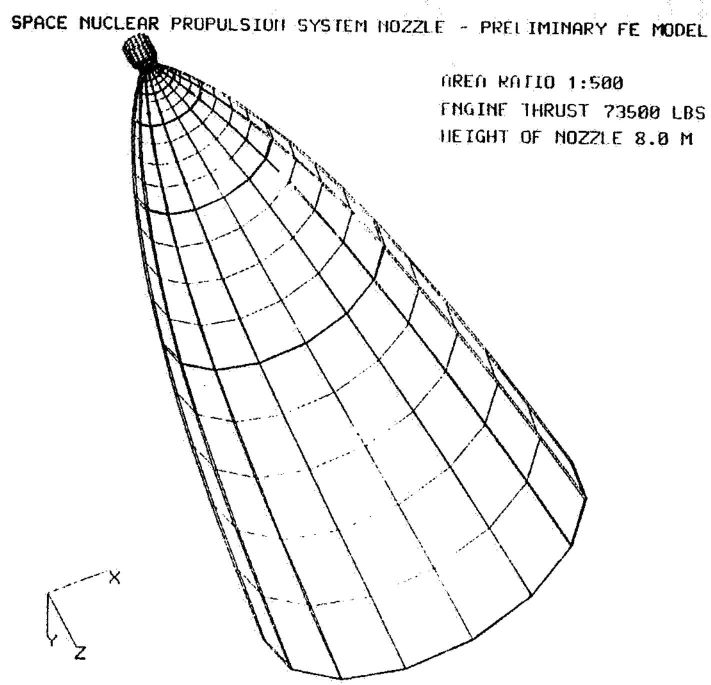
Dr. Ashwin Shah

Sverdrup Technology

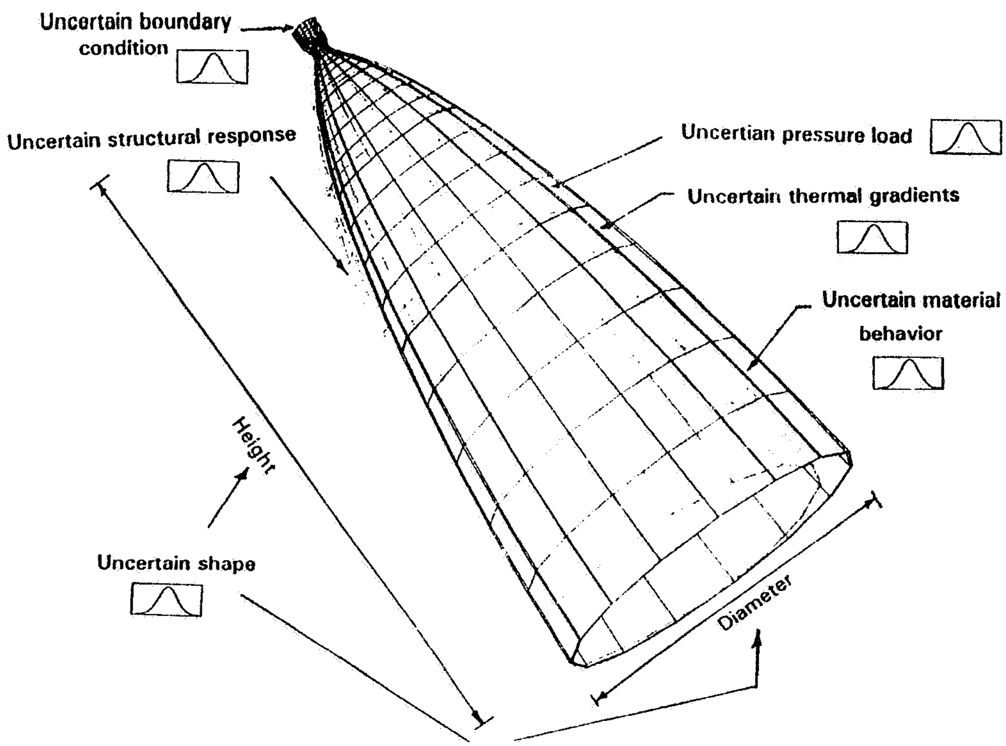
(presented by J.R. Stone, LeRC/NPO)

CERTIFICATION OF SPACE NUCLEAR PROPULSION SYSTEM NOZZLE WITH ASSURED RELIABILITY

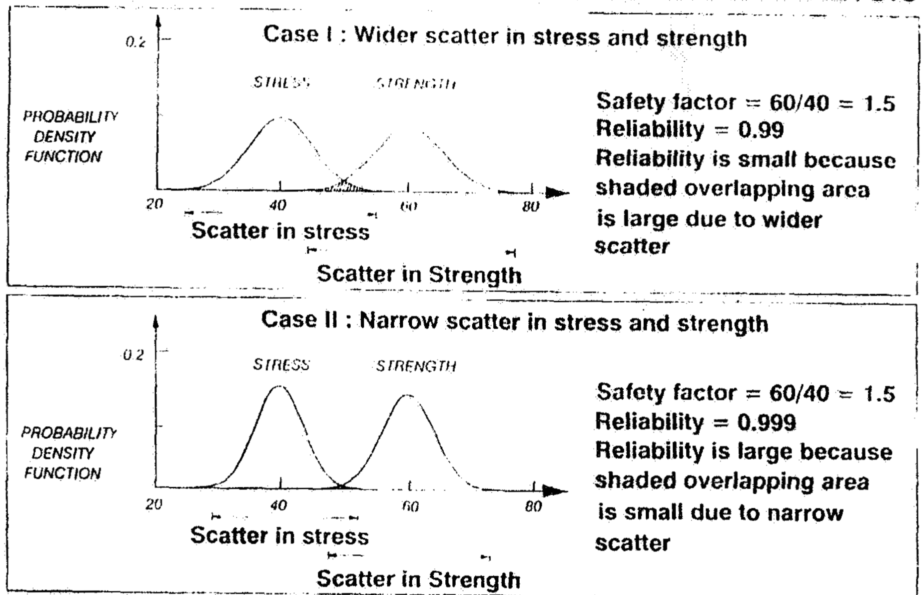
OBJECTIVE: To develop a methodology to certify Space Nuclear Propulsion System Nozzle with assured reliability



Certification of Space Nuclear Propulsion System Nozzle with Assured Reliability



ADVANTAGE OF PROBABILISTIC STRUCTURAL ANALYSIS

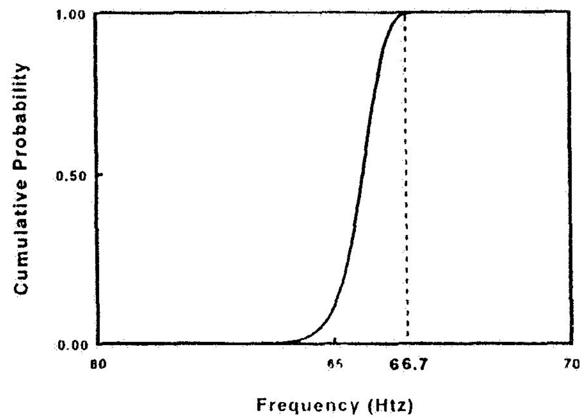


Reliability of structure depends on scatter in stress and strength
Probabilistic approach accounts for scatter in stress and/or strength rationally

Space Nuclear Propulsion System Nozzle
Uncertainties in the Random Variables

Random Variable		Coefficient of Variation /Standard Deviation	Distribution
Pressure		5 %	Normal
Geometry: X- Coordinate		0.25 In	Normal
Geometry: Y-Coordinate		0.25 In	Normal
Geometry: Z-Coord. (Height)		0.25 In	Lognormal
Thickness		2.5 %	Normal
Temperature Gradient	Inside surface	5 %	Normal
	Layer 2	5 %	Normal
	Layer 3	5 %	Normal
	Layer 4	5 %	Normal
	Outside surface	5 %	Normal
Modulus of Elasticity		5 %	Weibull
Coefficient of thermal Expansion		2.5 %	Normal
Strength		4 %	Weibull

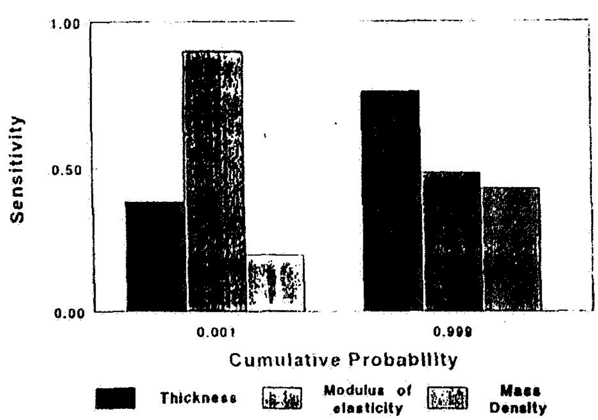
SNPS nozzle natural frequency



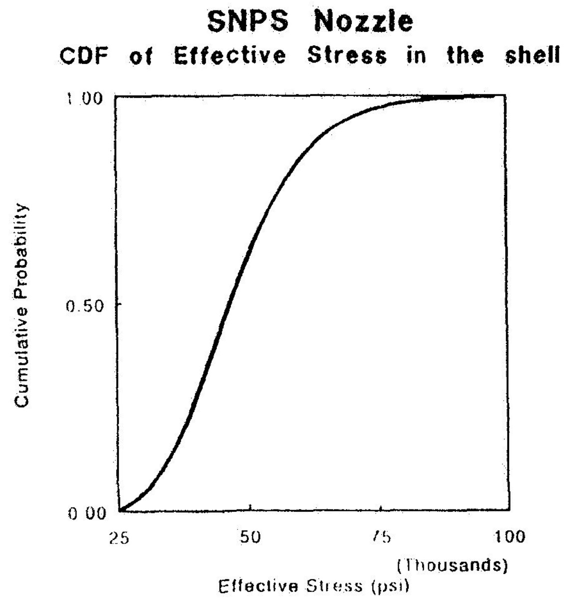
Probability of the natural frequency being less than 66.7 Hz = 0.999

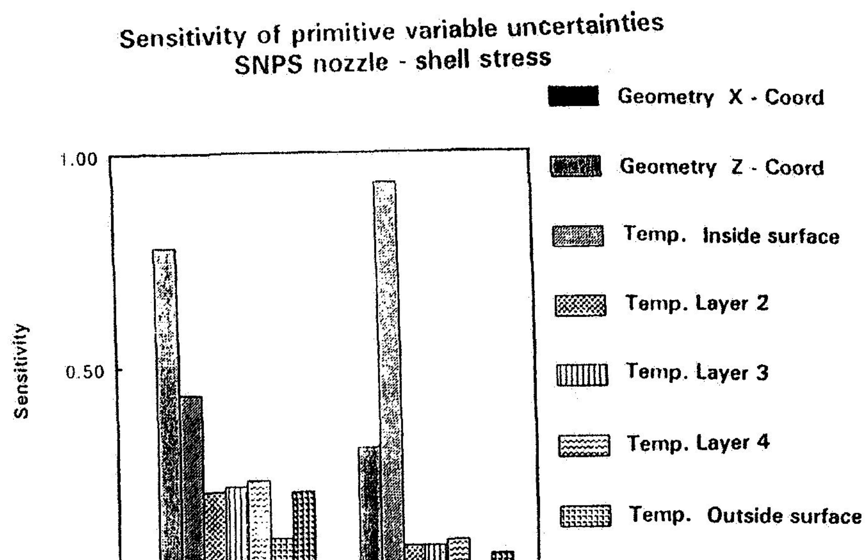
Therefore, to achieve a reliability of 0.999, the frequency of exciting force should be larger than 66.7 Hz to avoid resonance.

Sensitivity of primitive variable uncertainties
SNPS nozzle natural frequency



Variabes controlling the scatter of natural frequency within 66.7 Hz are thickness, modulus of elasticity and mass density. Therefore a tighter tolerance for the thickness and material properties are essential.

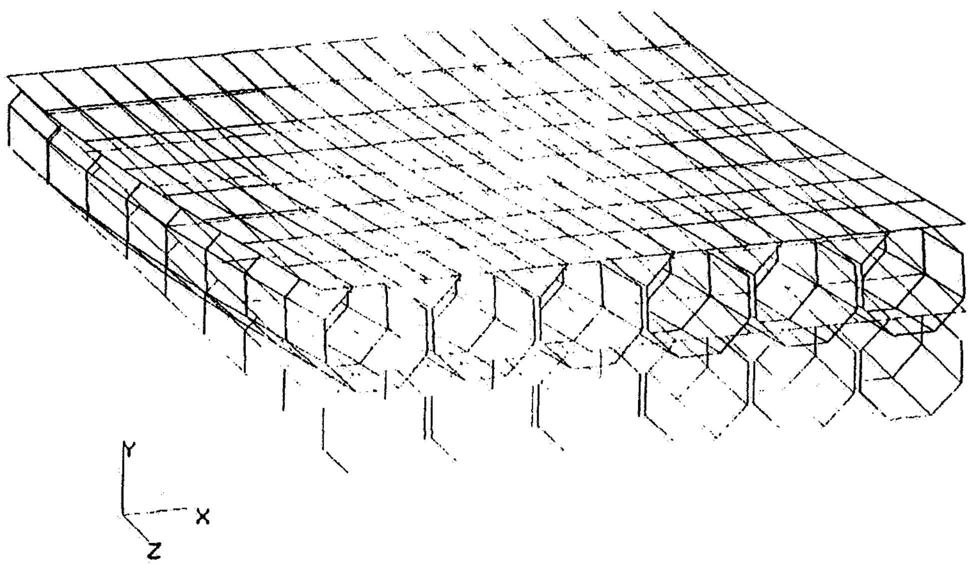




To control the stresses in the shell and achieve higher reliability, the uncertainties in the inside surface temperature should be reduced.

Work in progress:

- Modelling of NERVA base model with coolant tubes
- Development of pseudo-super element to reduce the size of the gloabal model to achieve computational speed and accuracy



N93-26932

MSFC NUCLEAR THERMAL PROPULSION TECHNOLOGY PROGRAM

MSFC

NON-NUCLEAR MATERIALS ASSESSMENT

> "IDENTIFY AND EVALUATE CANDIDATE MATERIALS FOR USE IN NTP TURBOMACHINERY AND PROPELLANT FEED SYSTEM APPLICATIONS."

THE APPROACH WAS TO DEVELOP AND IMPLEMENT DETAILED TEST PLANS AND EVALUATION CRITERIA FOR EFFECTS OF A HOT HYDROGEN ENVIRONMENT ON NTP CANDIDATE MATERIALS.

THE FOLLOWING MATERIALS WERE SELECTED FOR SURFACE EROSION TESTING:

- INCONEL 718 (BASELINE MATERIAL)
- IMAR 246 (IMPROVED MARAGING STEEL)
- NASA 23 (NICKEL BASED STEEL)

THE FOLLOWING TEST PLANS WERE DEVELOPED:

- STATIC HOT HYDROGEN TESTING UP TO 1000 C AND 5000 psi. TEST TO INCLUDE:
  - TENSILE PROPERTIES
  - LOW CYCLE FATIGUE
  - CREEP
  - FRACTURE TOUGHNESS
- FLOWING HOT HYDROGEN TESTING AT TEMPERATURES UP TO 1000 C FOR MICROSTRUCTURE CHARACTERIZATION OF EXPOSED MATERIALS

MATERIAL SAMPLES WERE TESTED AT AUBURN UNIVERSITY IN A 700 C HYDROGEN ENVIRONMENT WITH ADDITIONAL TESTS TO BE PERFORMED AT MSFC, ALSO AT 700 C.

MSFC FACILITY IS CAPABLE OF MATERIAL EXPOSURE TESTING UP TO 980 C AND 5000psi IN A HYDROGEN OR HELIUM ENVIRONMENT.

NON-NUCLEAR MATERIALS ASSESSMENT

The objective of the MSFC materials effort is to identify and evaluate candidate materials for use in NTP turbomachinery and propellant feed system applications. The initial task was to develop a set of test plans and evaluation criteria that could be applied to screen candidate materials for application in NTR components. In order to be a viable candidate, the material must be resistant to degradation due to the effects of exposure to hydrogen. A set of baseline materials were selected which included Inconel 718, NASA 23, and IMAR 246. These material samples were provided to Auburn University for exposure in a 700 C hydrogen environment and characterization of the induced surface erosion. Similar hydrogen environment testing was performed at MSFC for obtaining the mechanical properties of the samples through in situ testing in 10 MPA (1500 psi) hydrogen at 700 C. In situ test capabilities include tensile strain properties, fatigue, crack growth, fracture toughness, creep, and four point bend.

NON-NUCLEAR MATERIALS ASSESSMENT

HIGH TEMPERATURE TESTING OF VARIOUS CARBIDE BASED COATINGS FOR APPLICATION TO TURBOPUMPS, TURBINE BLADES, FLOW SYSTEMS, AND NOZZLES HAVE ARE BEING PERFORMED.

TaC, WC, & NbC HAVE BEEN EXPOSED TO HYDROGEN AT 1 ATMOSPHERE AND TEMPERATURES OF 830, 1350, & 1460 C TO DETERMINE % WEIGHT LOSS

Material	% weight loss at temperature 830 C	1350 C	1460 C
TaC	0.03	0.03	0.06
WC	0.07	0.10	0.04
NbC	0.10	0.36	1.14

SILICON NITRIDE AND ALUMINA CERAMICS HAVE BEEN TESTED FOR HIGH TEMPERATURE COATING APPLICATIONS. COMPARATIVE TESTING WAS PERFORMED IN AIR AND HYDROGEN AT AMBIENT TEMPERATURE AND AT 700 C FOR PERIODS OF 1 HOUR.

A FOUR-POINT BEND FIXTURE WAS USED TO TEST BASIC MECHANICAL PROPERTIES OF MEAN STRENGTH AND WEIBULL MODULUS.

NON-NUCLEAR MATERIALS ASSESSMENT

High temperature testing of carbide based coatings for application to turbopumps, turbine blades, flow systems, and nozzles are being performed. These coatings include the carbides of Tantalum, Niobium, Tungsten, and Silicon. These materials are being exposed to elevated temperatures over the range of 830 C to 1460 C in a vacuum and hydrogen environment for periods of one hour. Analysis consists of in situ weight loss determinations and residual gas analysis with subsequent examination of microstructure via Scanning Electron Microscopy and Transmission Electron Microscopy. The objective of this investigation is to characterize microstructural changes in these materials as a result of exposure to hydrogen.

Further material evaluations involve the preparation of Silicon Nitride and Alumina, candidate ceramics for high temperature coatings, for comparative tests in air and hydrogen at room temperature and 700 C for periods of one hour. At MSFC a four point bend fixture was configured to perform in situ testing of these materials for determination of their basic mechanical properties of mean strength and Weibull modulus.

MSFC

NTP TURBOMACHINERY TECHNOLOGIES

"DEVELOP AND VALIDATE ADVANCED TURBOMACHINERY TECHNOLOGIES AT THE COMPONENT AND TURBOPUMP ASSEMBLY LEVELS."

THE NASA REFERENCE SIZE WAS BOUNDED BY 50K AND 75K LB THRUST ENGINE.

THE APPROACH USED WAS TO ASSESS AND DEFINE TURBOMACHINERY TECHNOLOGY REQUIREMENTS FOR A SPACE BASED/START NUCLEAR THERMAL ROCKET ENGINE. MSFC AND LeRC SPECIALISTS COLLABORATED TO DEFINE AN INITIAL TECHNOLOGY PLAN FOR TPA TECHNOLOGY.

THE PLAN ADDRESSES:

- BEARINGS (FLUID FILM & FOIL)
  - SLOW START TRANSIENT FOR FLUID FILM BEARINGS
  - RUB TOLERANT MATERIALS FOR FLUID BEARINGS
  - ROLLING ELEMENT CAGE MATERIAL FOR THRUST BEARING (IF REQUIRED)
  - MAGNETIC BEARINGS
- SEALS
  - DEFINITION OF SEAL REQUIREMENTS
  - RUB TOLERANT MATERIALS
- EARLY NEED FOR A PROPELLANT FEED SYSTEM TEST BED
  - EARLY DEFINITION OF TRANSIENT LOADS
  - EVALUATION OF FEED SYSTEM IMPACTS ON TURBOMACHINERY

NTP TURBOMACHINERY TECHNOLOGIES

The objective of the MSFC turbomachinery technology task is to develop and validate technologies at the component and turbopump assembly level for application in a Nuclear Thermal Rocket engine. Marshall Propulsion Laboratory personnel collaborated with turbomachinery specialists at LeRC on the assessment of the technology requirements and priorities as well as an initial technology development plan. The ground rules provided that the engine size be in the range of 50K to 75K lb thrust and space based. Space base/start imposes a need to assess the requirement for low NPSH technologies.

The technology assessment and development plan addresses both fluid film and foil bearings. Current thinking is that rolling element bearings would not be used unless in a thrust bearing application. There exist, to date, little experience with either foil or hydrostatic bearings. Most experience addresses only fast start transient systems and. therefore, indicate a need for research in the application of fluid film/foil bearings in a prolonged start transient such as the NTR application. This also illustrates the need for further materials research for materials that would be wear resistant in a hydrogen and radiation environment. The main concern for rolling element bearings is for application as a thrust bearing where research is needed to identify a cage material that will survive the radiation environment. The application of a magnetic bearing could eliminate wear at startup and is also being considered.

Additional technology is also required in the area of seals. Questions must be addressed as to the need for a positive seal between the pump and turbine for pre and post operation.

A propellent feed system test bed is needed early in any TPA technology/advanced development program. A preliminary study has begun to assess the possibility of using existing turbomachinery and test stand hardware to facilitate the development of a test stand. The testbed is needed to evaluate transient operation and provide early definition of transient loads. This facility could also be used to assess feed system impacts on the turbomachinery. The system could also be used to evaluate TPA control and health monitoring technologies.

MSFC

HIGH TEMPERATURE SUPERCONDUCTING MAGNETIC BEARING TECHNOLOGY

"DEVELOP AND VALIDATE ADVANCED TECHNOLOGY FOR HIGH TEMPERATURE SUPERCONDUCTOR (HTS) PASSIVE MAGNETIC/HYDROSTATIC BEARING"

- GREATLY REDUCE, OR ELIMINATE COMPLETELY, THE EXPECTED WEAR TO A CONVENTIONAL HYDROSTATIC BEARING AS A RESULT OF NTR SLOW STARTUP TRANSIENT
- SDIO CONTRACT WITH MTI JOINTLY FUNDED BY DARPA AND NASA
- HTS TECHNOLOGY WILL ENABLE THE DEVELOPMENT OF A NEAR ZERO-WEAR BEARING WHEN COMBINED WITH FLUID FILM BEARING CONCEPTS.
- CURRENTLY DESIGNING PROOF-OF-CONCEPT TEST RIG BASED ON MSFC SUPPLIED REFERENCE TPA PARAMETERS BASED ON J-2S TURBOMACHINERY.
- TESTS AND MATERIAL RESEARCH ONGOING TO INCREASE HTS BEARING LOAD CARRYING CAPABILITY AT LH2 TEMPERATURES. HTS LOAD CAPABILITY HAS BEEN IMPROVED BY 450%.
- MSFC INHOUSE EFFORTS ARE FOCUSED ON MEASUREMENT OF HTS MAGNETIZATION OVER TEMPERATURE RANGE FROM 25K TO 77K. NTP-TPA OPERATIONAL TEMPERATURE IS PREDICTED TO BE AROUND 30K.

HIGH TEMPERATURE SUPERCONDUCTING MAGNETIC BEARING TECHNOLOGY

The objective of the MSFC HTS technology task is to develop and validate advanced technology for High Temperature Superconductor (HTS) passive magnetic/hydrostatic bearings for application in a nuclear rocket engine. This bearing concept will greatly reduce, or eliminate completely, the expected wear to a conventional hydrostatic bearing as a result of the extremely slow startup transient of the NTR. This work was performed by Mechanical Technology Inc. under a SDIO contract funded jointly between NASA/MSFC and DARPA.

By combining HTS technology with that of fluid film bearings, it will be possible to suspend the pump shaft during the start-up and shut-down of the pump when the hydrostatic bearing is not fully functional. HTS stiffness has been improved by 450% during the course of this effort and further improvement to capabilities of >2000 lb/in/in2 is believed very possible. MTI was supplied reference parameters based on the J-2S turbopump in order to design a proof-of-concept test apparatus.

Additional inhouse efforts have focused on measurement of HTS magnetization over temperature ranges from 25K to 77K. The operational temperature of the turbomachinery for the NTR is predicted to be 30K.

NUCLEAR GAS CORE PROPULSION RESEARCH PROGRAM

a presentation to the

Nuclear Propulsion Technical Interchange Meeting '92
NP-TIM-92

NASA Lewis Research Center
October 20 - 23, 1992

by

Nils J. Diaz, Principal Investigator
Edward T. Dugan, Co-Principal Investigator
Samim Anghaie, Co-Principal Investigator

INSPI

Innovative Nuclear Space Power & Propulsion Institute
University of Florida

Prepared under NASA Grant NAG3-1293

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NUCLEAR GAS CORE PROPULSION RESEARCH PROGRAM

Advanced Nuclear Propulsion Studies

- To develop a hydrogen properties package at temperatures 10 - 10,000 K and pressures 0.1 - 200 atm.
- To develop a transient simulation program for parametric studies and design analysis of high temperature nuclear rockets

Nuclear Vapor Thermal Rocket (NVTR) Studies

- To conduct nuclear and thermal design optimization of the NVTR fuel, fuel elements and core geometry
- To develop a system and parametric analysis code for the NVTR

Ultrahigh Temperature Nuclear Fuels and Materials Studies

- Determine properties of UF_4 and UF_4 mixtures nuclear fuels at temperature - pressure ranges of interest to advanced nuclear propulsion systems
- Measure/model high temperature compatibility of UF_4 with refractory carbides.

10/20/92

The objectives of these studies are to develop models and experiments, systems and fuel elements for advanced nuclear thermal propulsion rockets. The fuel elements under investigation are suitable for gas/vapor and multiphase fuel reactors.

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EVALUATION OF PARA- AND DISSOCIATED HYDROGEN PROPERTIES AT T = 10 - 10,000 K

- NASA/NIST Property Package
  (13.8 < T < 10,000 K and .1 < P < 160 bar)
  - Molecular Weight, Density
  - Enthalpy, Entropy
  - Specific Heats, Specific Heat Ratio
  - Thermal Conductivity, Viscosity
- Hydrogen Property Generator Code Features
  - Linear Interpolation
  - Natural Cubic Spline
  - Least Square Curve Fitting with Pentad Spline Joint Functions
- Graphical Representation of Properties

10/20/92

The hydrogen property generator utilizes two interpolation techniques and a least-square curve fitting routine with a pentad spline function which links least-square fitted pieces together. The property generator package is incorporated into the NTR simulation code and also into a system of CFD-HT codes.

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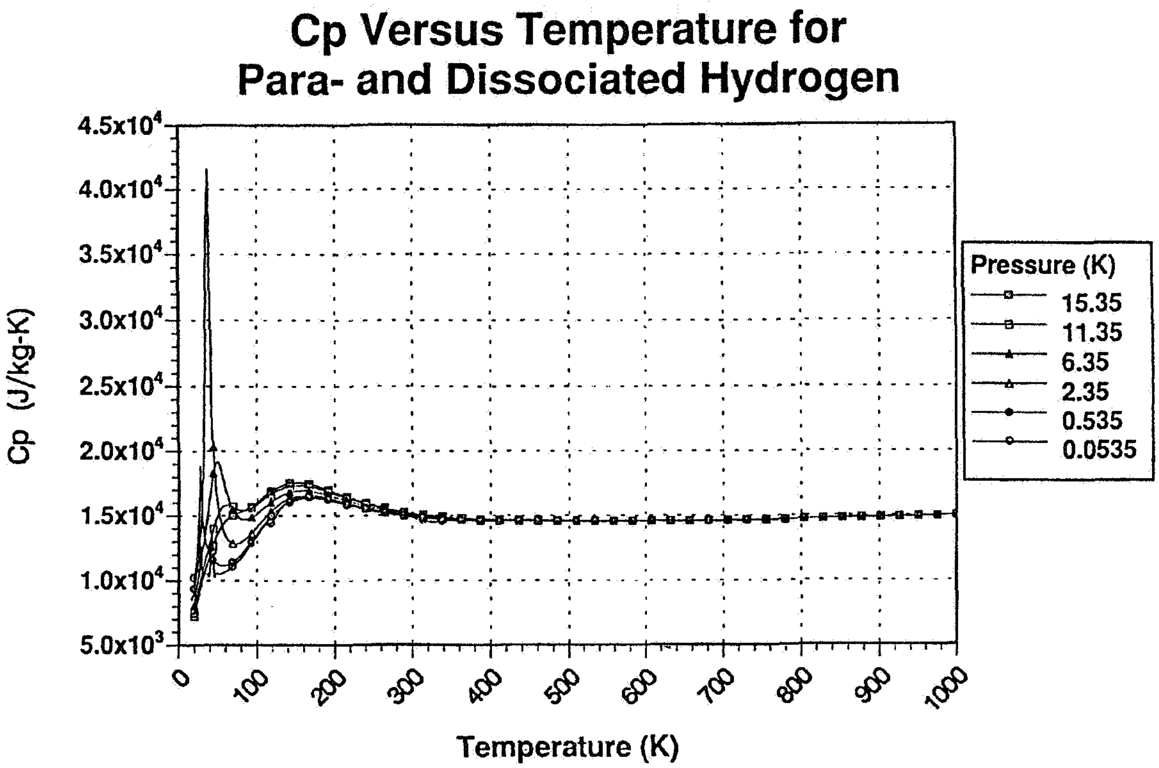

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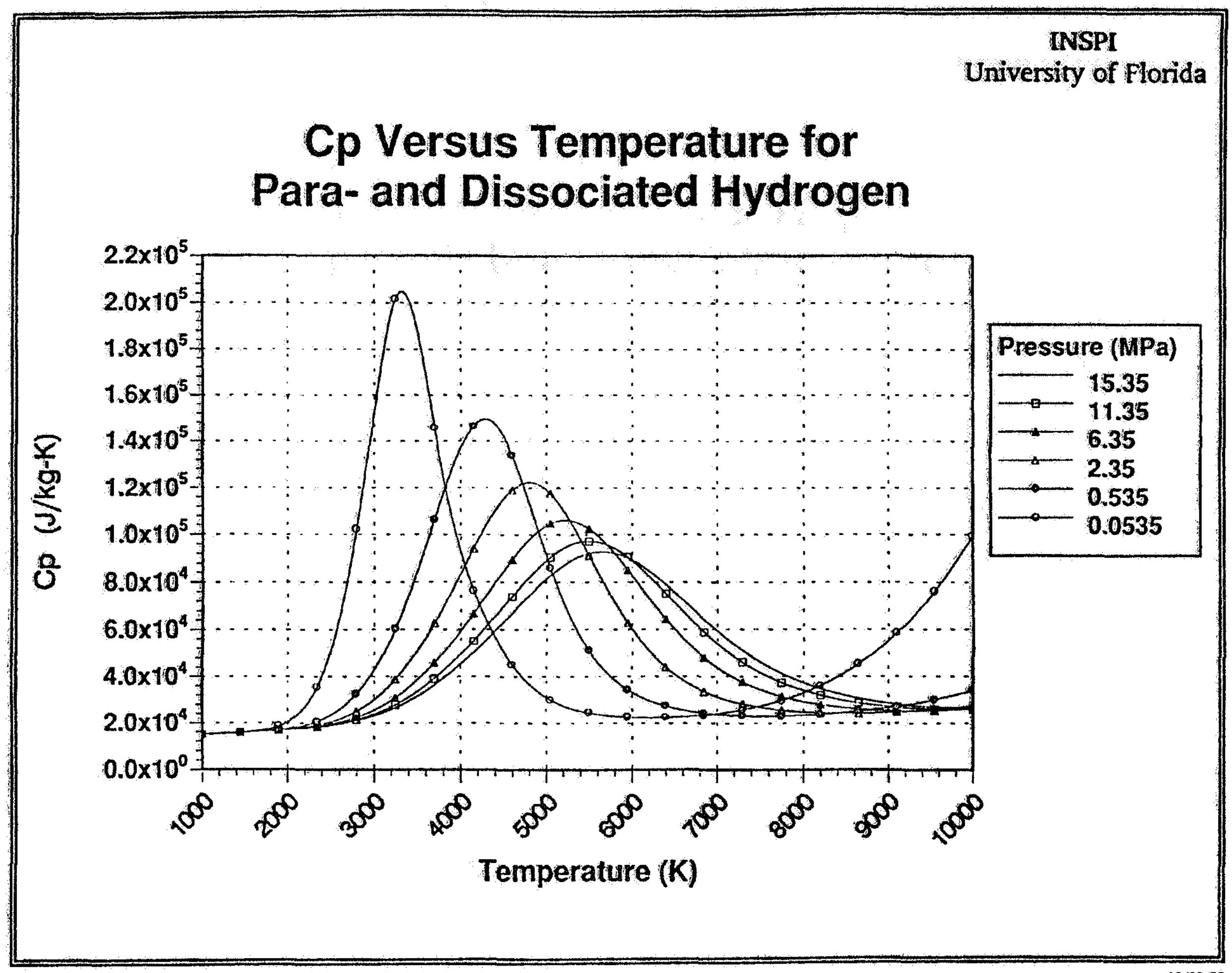
Heat capacity of hydrogen near the critical point shows large gradient and oscillatory behavior. At p = 2.35 MPa the property package indicates a sharp peak for C_p.



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At higher temperatures, the heat capacity data displays smooth behavior. The sharp increase in C_p value at temperatures above 2000 K is due to hydrogen dissociation.

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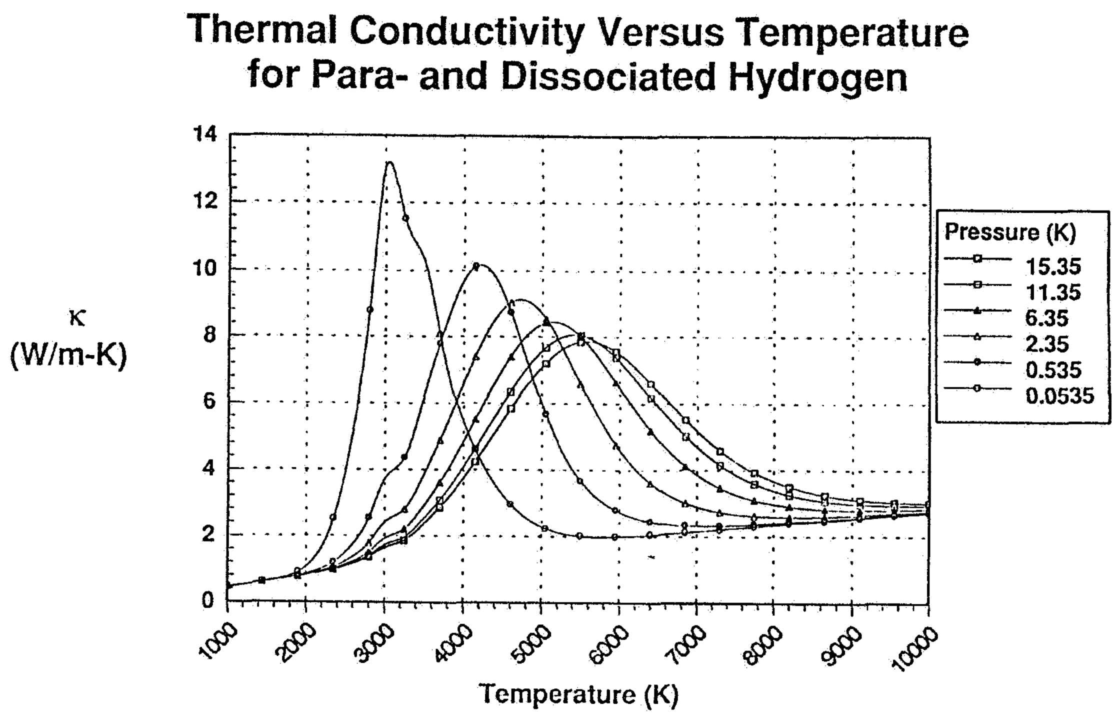

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The hydrogen property package is a combination of two subpackages covering the temperature ranges 10 - 3000 K and 3000 - 10,000 K, respectively. The large change of gradients in hydrogen viscosity at 3000 K indicates a non-physical flaw in the model.

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MODELING AND ENGINEERING SIMULATION OF NUCLEAR THERMAL ROCKET SYSTEMS

- Modular Thermal Fluid Solver with Neutronic Feedback
- Main Component Modules:
  - Pipes, Valves, Mixer
  - Nozzle Skirt
  - Pump, Turbine
  - Reflector, Reactor Core
- Hydrogen (Para- and Dissociated) Property Package
  - $10 \leq T \leq 10{,}000$ K
  - $.1 \leq P \leq 160$ bar
- Models Developed for NTVR, NERVA and XNR 2000
- CFD and Heat Transfer Models for Main NTR Components

10/20/92

A detailed program for modeling of full system nuclear rocket engines is developed. At present time, the model features the expander cycle. Axial power distribution in the reactor core is calculated using 2- and 3-D neutronics computer codes. A complete hydrogen property model is developed and implemented. Three nuclear rocket systems are analyzed. These systems are: a 75,000 lbf NERVA class engine, a 25,000 lbf cermet fueled engine and INSPI's nuclear thermal vapor rocket.

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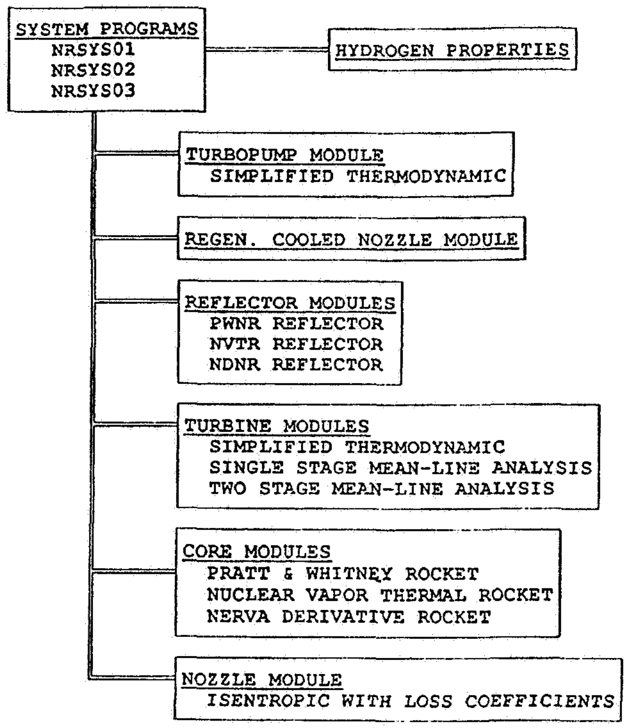
NUCLEAR THERMAL ROCKET SIMULATION SYSTEM



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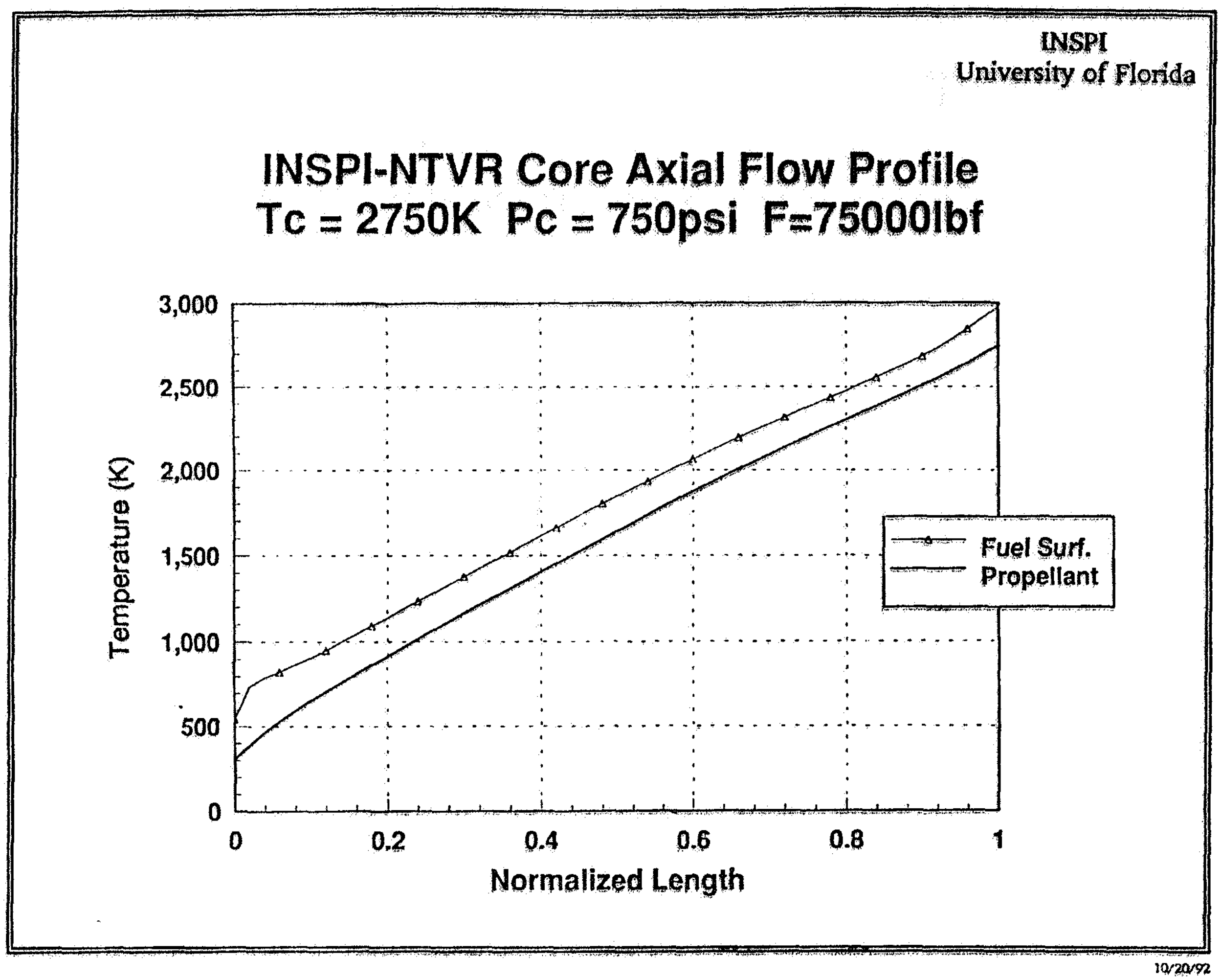
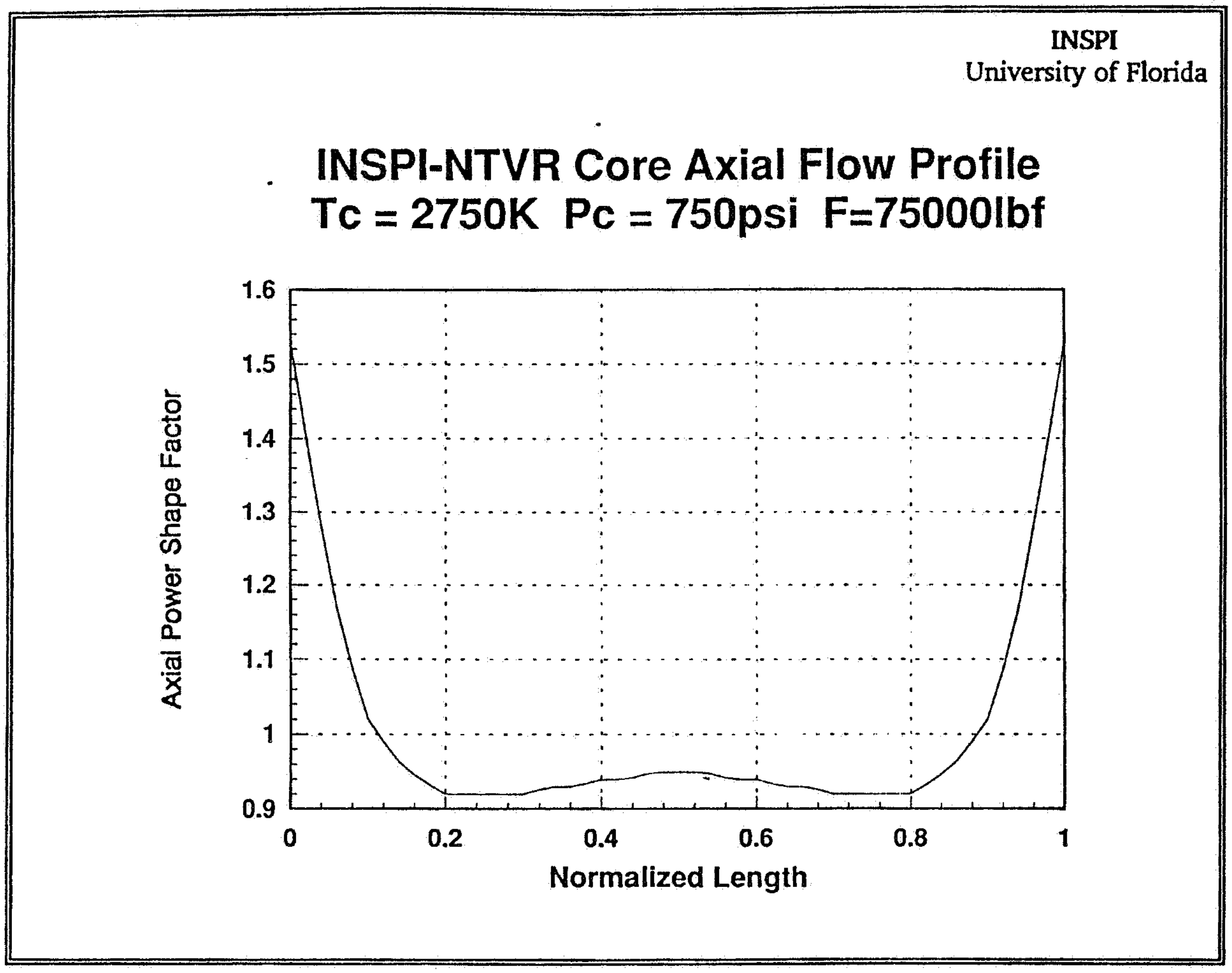
The main program links all the component modules and iterates to arrive at the user specified thrust chamber pressure and temperature and thrust level. Reactor power and propellant flow rate are among outputs of the simulation program. Fuel elements in the core module are prismatic with variable flow area ratio. Each module divides the relative component into N segments.



Axial temperature distribution of NVTR fuel surface and propellant in an average power rod. Reactor power is adjusted to achieve the thrust chamber temperature and pressure of 2750 K and 750 psi, respectively.



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Normalized axial power distribution in C-C composite fuel matrix NTVR, calculated by DOT-2 S_n code. The axial power shape factor is an input for the simulation code.

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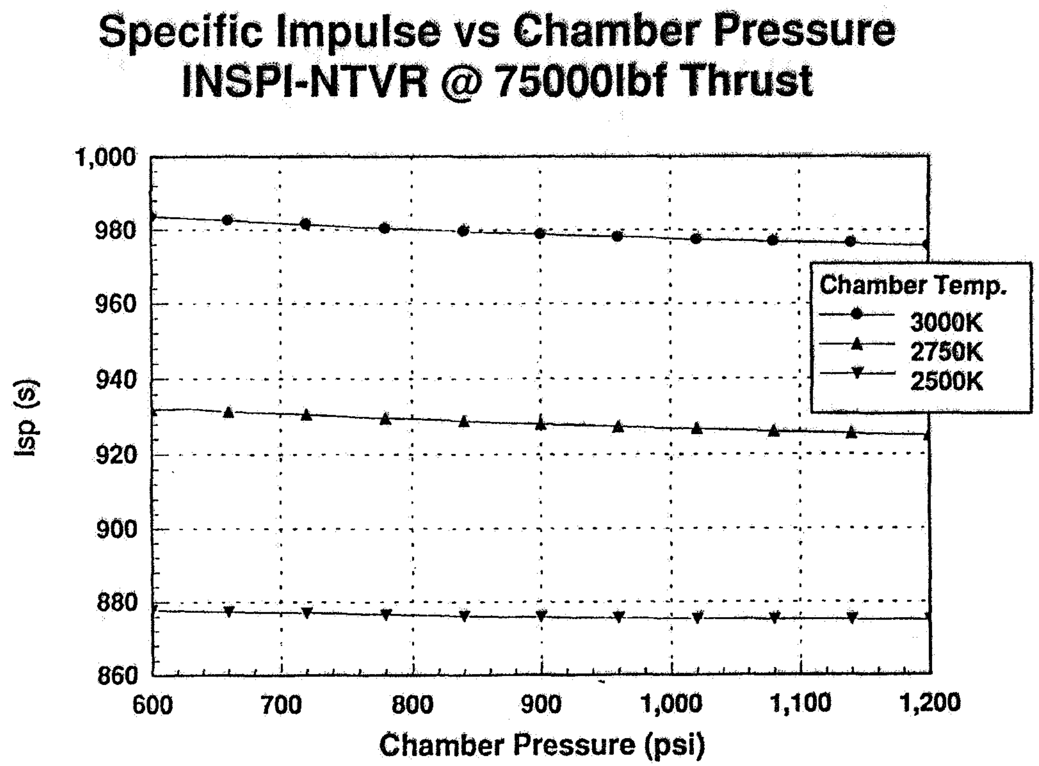

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Parametric study of thrust chamber pressure and temperature impact on Isp of NTVR. At higher pressures Isp is less sensitive to thrust chamber temperature.

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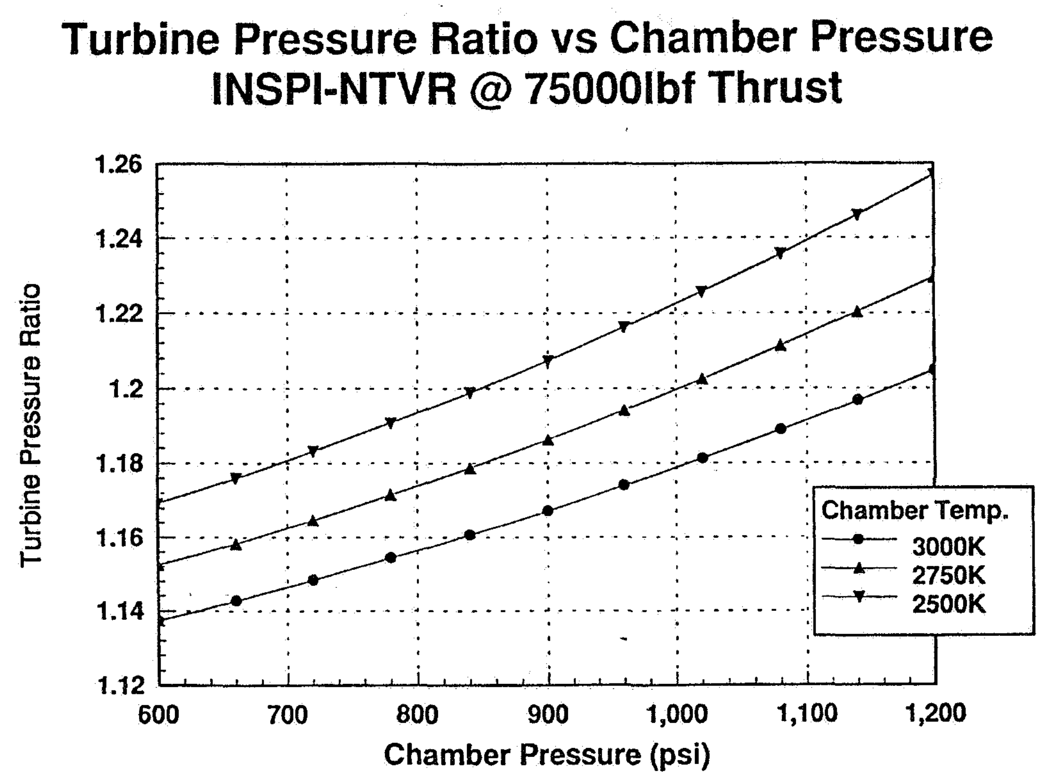

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Turbine pressure ratio is sensitive to both thrust chamber pressure and temperature. For thrust chamber pressure of 1200 psi and temperature of 3000 K, the turbine pressure ratio of 1.26 is well within the range of available technology.

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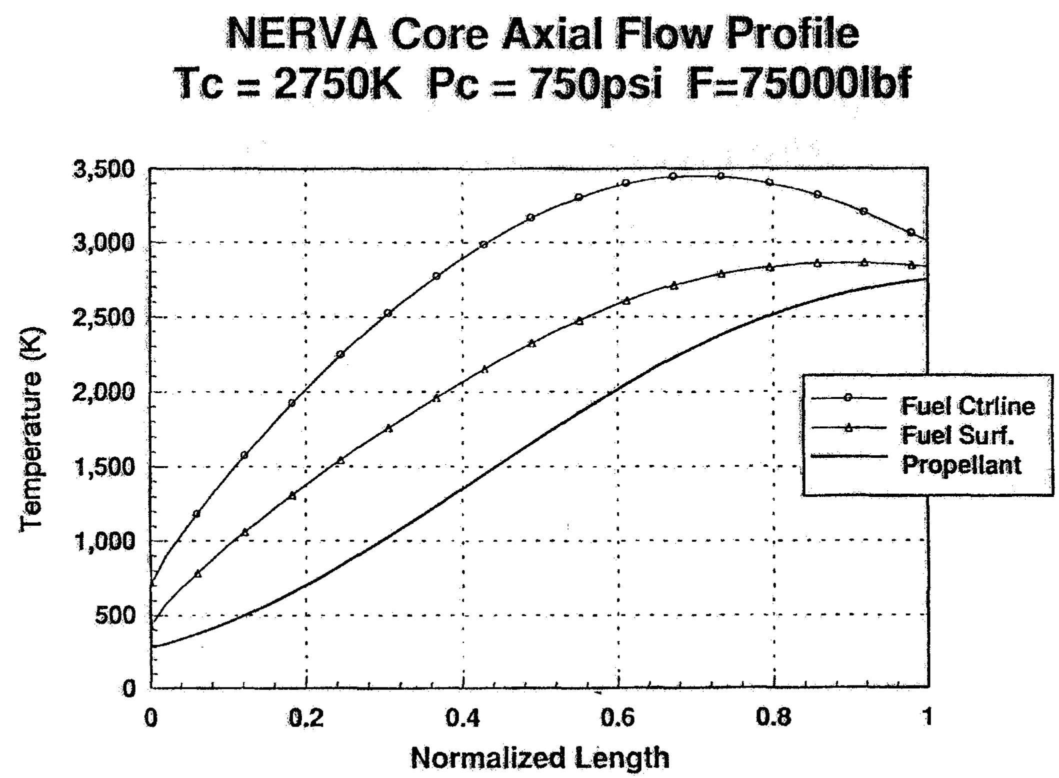

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Axial temperature profiles for NERVA-75,000 lbf engine are presented. The maximum fuel temperature is 3490 K at .7 m from the core entrance.

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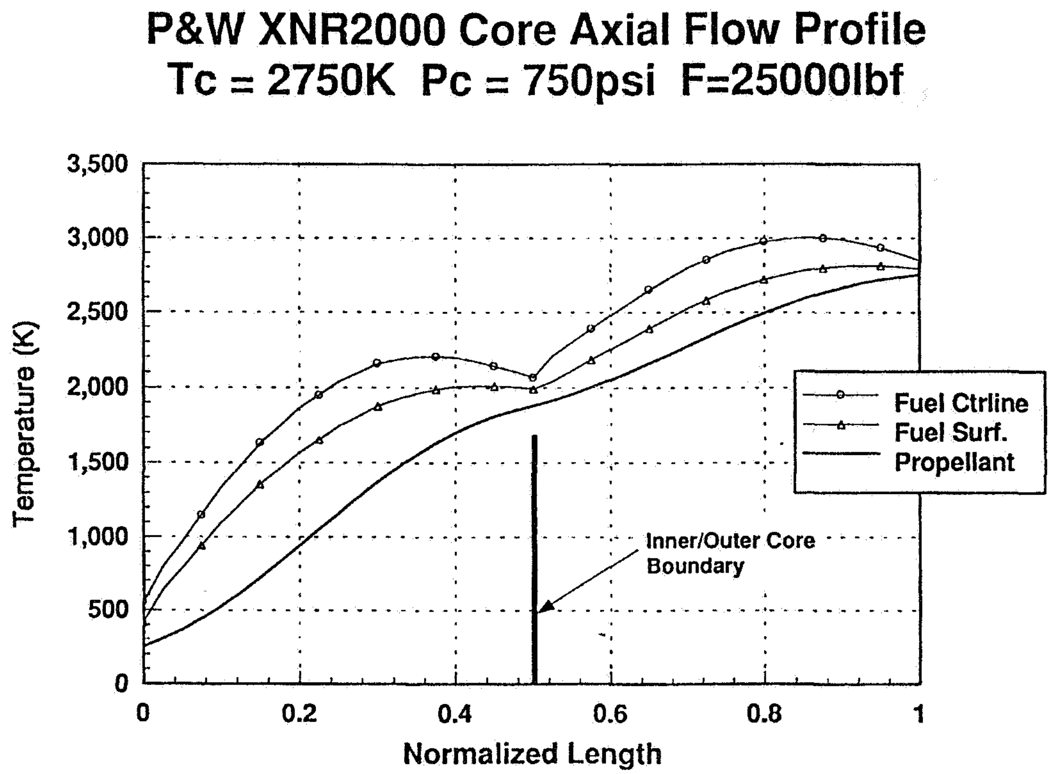

10/20/92

Axial temperature distribution in XNR 2000 core is presented. XNR 2000 features a two path folded flow core fueled with CERMET. The maximum fuel temperature is 3000 K at about 85% from the entrance to the inner core region.

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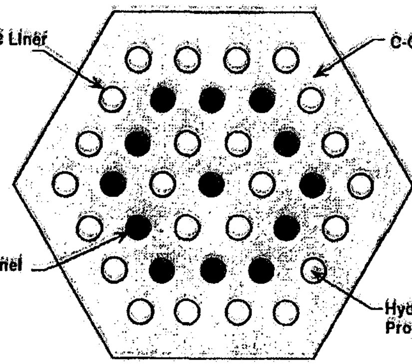
NUCLEAR VAPOR THERMAL ROCKET (NVTR)

NVTR PERFORMANCE

Thrust = 75,000 lbs
Isp = 1000 sec
H flow = 30 kg/sec
H TEMP = 3100 K
T/W = 5

CORE: 2000 fuel element
Radius = 0.5 m
Height = 1.5 m
Fuel channel/ele. = 24-32
H channel/ele. = 24-32
Critical mass = 20 kg
H pressure = 100 atm
UF_4 pressure = 100 atm
Fuel Center T = 4500 K

FWD Reflector (BeO) = 15 cm
Aft Reflector (C-C Composite) = 25 cm
Radial Reflector (BeO) = 15 cm



Schematic of NVTR Fuel Element

10/20/92

The Nuclear Vapor Thermal Rocket (NVTR) is an advanced thermal propulsion engine, using vapor or multiphase nuclear fuel, with predicted performance at the upper limits of solid core reactors. The NVTR also serves as base technology development toward high performance Gas Core Reactors.

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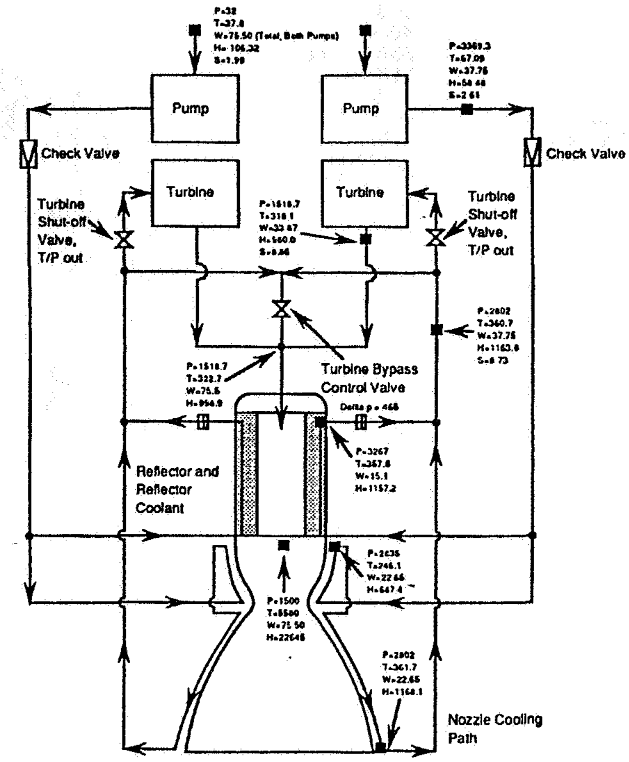
NUCLEAR VAPOR THERMAL ROCKET (NVTR)
75 K-lbf NVTR, Expander Cycle, Dual T/P



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Design Values

Pump Flowrate (Total)	75.50	lb_m/sec
Pump Discharge Pressure	3,369	psia
Number Of Pump Stages	2	
Pump Efficiency (%)	78.26	%
Turbopump Rpm	70,000	RPM
Turbopump Power (Each)	8,802	HP
Turbine Inlet Temp	361	deg-R
Number Of Turbine Stages	2	---
Turbine Efficiency	81.51	%
Turbine Pressure Ratio	1.85	---
Turbine Flow Rate (Each)	33.87	lb_m/sec
Reactor Thermal Power	1,759	MW
Fuel Element Transferred Power	1,724	MW
Nozzle Chamber Temperature	5,580	deg-R
Chamber Pressure (Nozzle Stagnation)	1,500	psia
Nozzle Expansion Area Ratio	500:1	---
Nozzle Percent Length	123	%
Vacuum Specific Impulse (Delivered)	993.3	sec

Heat loads are as follows:

Nozzle-con (total):	30.05	MW
Nozzle-div (total):	22.97	MW
Reflector (total):	35.00	MW

P = psia
T = deg-R
W = lb_m/sec
H = BTU/lb_m
S = BTU/lb_m-R

Note: Flows indicated are for one-half of system

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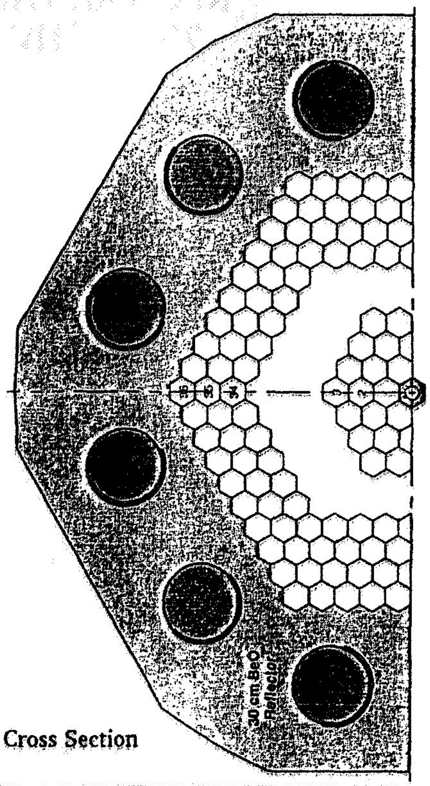
NUCLEAR VAPOR THERMAL ROCKET PARAMETERS

Fuel Pressure	100 atm
Average Fuel Temperature	4000 K
Maximum Element Heat Flux	330 W/cm^2
Nominal Element Length	150 cm
Fuel Volume Fraction	0.15
Coolant Volume Fraction	0.15
Moderator Volume Fraction	0.70
Fuel Element Power	0.7 MWt
Element Heat Transfer Area	2170 cm^2
Reactor Core L/D	1.5
Fuel Channel Diameter	0.142 cm
Fuel Channel Sectional Area	0.0158 cm^2
Total Fuel Channel Area Per Element	0.505 cm^2
Fuel Element Sectional Area	3.464 cm^2
Element Diameter (across flats)	2.00 cm
Coolant Channel Diameter	0.142 cm
Coolant Channel Sectional Area	0.0158 cm^2
Total Coolant Channel Area Per Ele.	0.505 cm^2
Core Volume (elements only)	1.053 m^3
Core Power Density	1330 MW/m^3
Fuel Element Mass, Total	1.35 MT
Forward Reflector Mass	0.60 MT
Aft Reflector Mass	0.51 MT
Radial Reflector Mass	2.47 MT
Radiation Shield Mass	0.90 MT
Total Reactor Mass	5.83 MT
Misc Engine Components Mass	0.9 MT
Total Engine Mass	6.83 MT
Engine F/W	5.0 MT



NVTR Cross Section

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HIGH TEMPERATURE NUCLEAR FUELS AND MATERIALS STUDIES

- Experimental Studies Related to a Parallel Program Confirmed UF_4 Compatibility with:

  - W at temperatures up to 2200 K
  (Experiment and post-test analysis at T up to 3000 K in progress)

  - Mo at temperatures up to 2200 K
  (Experiment and post-test analysis at T up to 2600 K in progress)

  - C at temperatures up to 1800 K

- Detailed Thermodynamics Analysis Demonstrated Outstanding Chemical Compatibility Between UF_4 and WC, W_2C, Mo_2C at Temperatures up to 2600 K

- Thermodynamic Studies Revealed Outstanding Properties of UF_4 - UC_2 Mixture for NTVR Fuel

10/20/92

Compatibility of UF_4 at elevated temperatures with wall materials is a key to successful development of fuel element for NTVR. Experimental studies of UF_4 compatibility with a wide range of materials has shown promising results for Mo, W, and C. Thermodynamic analysis suggested outstanding chemical compatibility of WC, W_2C and Mo_2C at temperatures up to 2600 K. High temperature thermodynamics analysis has also revealed the outstanding stability of UF_4 - UC_2 system. Due to presence of carbon in UF_4 - UC_2 fuel mixture, better compatibility with the fuel element wall materials and gaseous fuel is expected.

N93-26934

Flow Instability in Particle-Bed Nuclear Reactors

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PRESENTED AT NUCLEAR PROPULSION INTERCHANGE MEETING
NASA LEWIS RESEARCH CENTER, OCTOBER 22, 1992

Abstract

The particle-bed core offers mitigation of some of the problems of solid-core nuclear rocket reactors. Dividing the fuel elements into small spherical particles contained in a cylindrical bed through which the propellant flows radially, may reduce the thermal stress in the fuel elements, allowing higher propellant temperatures to be reached. The high temperature regions of the reactor are confined to the interior of cylindrical fuel assemblies, so most of the reactor can be relatively cool. This enables the use of structural and moderating materials which reduce the minimum critical size and mass of the reactor. One of the unresolved questions about this concept is whether the flow through the particle-bed will be well behaved, or will be subject to destructive flow instabilities. Most of the recent analyses of the stability of the particle-bed reactor have been extensions of the approach of Bussard and Delauer, where the bed is essentially treated as an array of parallel passages, so that the mass flow is continuous from inlet to outlet through any one passage. A more general three dimensional model of the bed is adopted here, in which the fluid has mobility in three dimensions. Comparison of results of the earlier approach to the present one shows that the former does not accurately represent the stability at low Re. The more complete model presented here should be capable of meeting this deficiency while accurately representing the effects of the cold and hot frits, and of heat conduction and radiation in the particle-bed. It can be extended to apply to the cylindrical geometry of particle-bed reactors without difficulty. From the exemplary calculations which have been carried out, it can be concluded that a particle bed without a cold frit would be subject to instability if operated at the high temperatures desired for nuclear rockets, and at power densities below about 4 megawatts per liter. Since the desired power density is about 40 megawatts per liter, it can be concluded that operation at design exit temperature but at reduced power could be hazardous for such a reactor. But the calculations also show that an appropriate cold frit could very likely cure the instability. More definite conclusions must await calculations for specific designs.

* R.C. Maclaurin Professor of Aeronautics and Astronautics

Conclusions

Comparison of three quite different approaches to modeling the stability of the particle-bed reactor, all with consistent physical assumptions, shows that a complete linear stability such as that presented here is in fact necessary for reliable prediction of the stability of the particle-bed reactor. The approach, termed here the Parallel-Stream model, where the reactor is assumed to be composed of a series of channels coupled only at their inlets and outlets, does not accurately represent the stability at low Re, nor does it represent the effect of heat conduction in the bed.

The model termed here (perhaps somewhat naively) the Complete Model should be capable of accurately representing the effects of the cold and hot frits, and of heat conduction and radiation in the particle bed. It can be extended to apply to the cylindrical geometry of particle-bed reactors without difficulty.

From the exemplary calculations which have been carried out, it can be concluded that a particle bed without a cold frit would be subject to instability if operated at the high temperatures desired for nuclear rockets, and at power densities below about 4 megawatts per liter. Since the desired power density is about 40 megawatts per liter, it can be concluded that operation at design exit temperature but at reduced power could be hazardous for such a reactor. But the calculations also show that an appropriate cold frit could very likely cure the instability. More definite conclusions must await calculations for specific designs.

Acknowledgments

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The author wishes to acknowledge helpful discussions with Dr. John Clark of NASA Lewis, with Prof. Frank E. Marble, and with Mssrs. James Kalamas, David Suzuki, Timothy Lawrence and Jonathan Witter.

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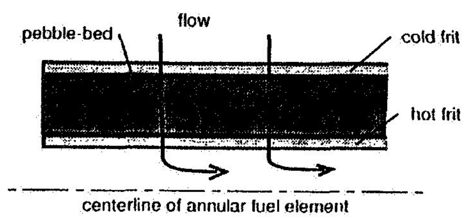
SCHEMATIC OF PARTICLE-BED



GOVERNING EQUATIONS

$$\nabla p = -\frac{\mu}{\kappa}\vec{u} - \frac{b}{\kappa}|\vec{u}|\,\rho\vec{u} \tag{1}$$

$$\frac{\mu}{\kappa} = 150\,\frac{(1-\varepsilon)^2}{\varepsilon^3}\,\frac{\mu(T_g)}{D_p^2}$$

$$\frac{b}{\kappa} = 1.75\,\frac{(1-\varepsilon)}{\varepsilon^3}\,\frac{1}{D_p} \tag{2}$$

$$\rho_s c_{ps}\frac{\partial T_p}{\partial t} = Q - \rho u c_p (T_p - T_g)\,h + k_{eff}\nabla^2 T_p \tag{3}$$

$$k_{eff} = k_{cond} + k_{rad} = k_{cond} + \frac{16}{3}\sigma D_p T_p^3 \tag{4}$$

$$h = \frac{75}{D_p}\,\frac{(1-\varepsilon)^2}{\varepsilon^2}\,\frac{1}{Re} + \frac{.875}{D_p}\,\frac{(1-\varepsilon)}{\varepsilon^2} \tag{5}$$

$$\rho c_p\frac{\partial T_g}{\partial t} + \rho\vec{u}\cdot\nabla c_p T_g = \rho u c_p (T_p - T_g)h \tag{6}$$

$$\frac{\partial \rho}{\partial t} + \nabla\cdot(\rho\vec{u}) = 0 \tag{7}$$

$$p = \rho R T \tag{8}$$

NON-DIMENSIONAL GOVERNING EQUATIONS

$$\nabla p = -b_1 T_g^{\nu} \vec{u} - b_2 |\vec{u}| \rho \vec{u} \tag{1n}$$

$$b_1 = \frac{\mu_0(0)\, u_0(0)\, l}{p_0(0)\, \kappa} \qquad b_2 = \frac{b\, \rho_0(0)\, u_0^2(0)\, l}{p_0(0)\, \kappa} \qquad \mu(T_g) = \mu_0(0)\left(\frac{T_g}{T_g(0)}\right)^{\nu}$$

$$c \frac{\partial T_p}{\partial t} = q - \rho u (T_p - T_g) H + K \nabla^2 T_p \tag{3n}$$

$$c \equiv \frac{\rho_s c_{ps}}{\rho_0(0)\, c_p}, \qquad H \equiv h\, l \qquad K \equiv \frac{k_{eff}}{\rho_0(0)\, u_0(0)\, c_p\, l} \equiv K_c + K_r T_p^3$$

$$q = \frac{Q\, l}{\rho_0(0)\, u_0(0)\, c_p T_0(0)}$$

$$\rho \frac{\partial T_g}{\partial t} + \rho \vec{u} \cdot \nabla T_g = \rho u (T_p - T_g) H \tag{6n}$$

$$\frac{\delta \rho}{\delta t} + \nabla \cdot \rho \vec{u} = 0 \tag{7n}$$

$$p = \rho T_g \tag{8n}$$

ZEROTH-ORDER OR STEADY SOLUTION

$$\rho_0 u_0 \frac{dT_{g0}}{dx} = q + K \frac{d^2 T_p}{dx^2}$$

$$T_{g0} = 1 + q x \tag{9}$$

$$T_{p0} = 1 + q x + \frac{q}{H} \tag{10}$$

$$p_0^2 = 1 - \frac{2 b_1}{(\nu + 2) q}\left[(1 + q x)^{\nu + 2} - 1\right] - \frac{b_2}{q}\left[(1 + q x)^2 - 1\right] \tag{11}$$

$$\rho_0 = \frac{p_0}{T_{g0}} \tag{12}$$

$$u_0 = \frac{1}{\rho_0} \tag{13}$$

FIRST ORDER

$$\nabla p = -b_1\left[T_{g0}^{\nu}\vec{u} + \hat{i}\,u_0\,\nu\,T_{g0}^{\nu-1}\,T_g\right] - b_2\left[\vec{u} + \hat{i}\left(u_0^2\,\rho + u_x\right)\right] \tag{14}$$

$$c\,\frac{\partial T_p}{\partial t} = -H(T_p - T_g) - H(T_{p0} - T_{g0})(\rho_0\,u_x + u_0\,\rho) + K\nabla^2 T_p \tag{15}$$

$$\begin{aligned} &\rho_0\frac{\partial T_g}{\partial t} + \rho_0\frac{dT_{g0}}{dx}u_x + u_0\frac{dT_{g0}}{dx}\rho + \frac{\partial T_g}{\partial x} = \\ &\rho_0(T_{p0}-T_{g0})\,H\,u_x + u_0(T_{p0}-T_{g0})\,H\,\rho + H\,(T_p - T_g) \end{aligned} \tag{16}$$

$$\frac{\partial \rho}{\partial t} + \rho_0\,\nabla\cdot\vec{u} + \frac{d\rho_0}{dx}u_x + u_0\frac{\partial \rho}{\partial x} + \frac{du_0}{dx}\rho = 0 \tag{17}$$

$$p = \rho_0\,T_g + T_{g0}\,\rho \tag{18}$$

Variables are: p, ρ, T_g, $\vec{u}$ and T_p

PARAMETERS.

Dimensionless Parameters :

from 1n, ν, b_1, b_2

from 3n, q, c, H, K_c, K_r

Operating Parameters :

T_{g0} (exit) = 3000 K

p_0 (exit) = 100 atm

$Q = 4\times10^{10}$ watt/m^3

Design Parameters :

l = 0.01 m

$D_p = .5\times10^{-3}$ m

Stability Parameters :

$$R_e = \frac{\rho_0(0)\,u_0(0)\,D_p}{\mu(0)} = \frac{D_p\,Q\,l}{c_p T_0\,\mu(0) q}$$

q .

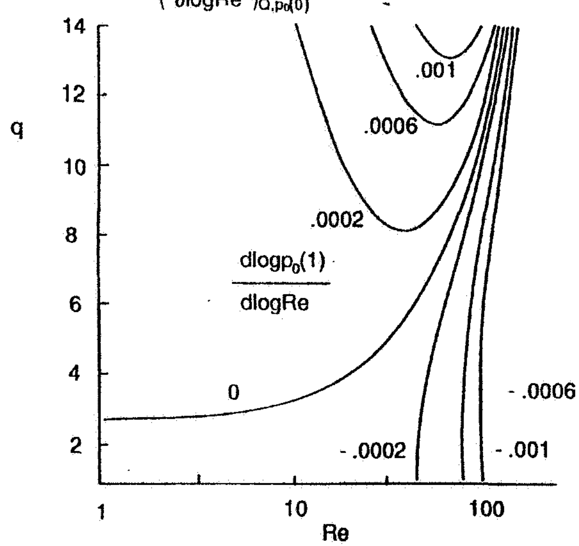
APPROACHES TO INSTABILITY ANALYSIS

1) Parallel Flow Instability
2) Local Instability Analysis
3) Full Stability Analysis

PARALLEL-STREAM INSTABILITY

Instability is possible if $p_O(1)$ increases with mass flow density for fixed Q and $p_O(0)$. Hence:

$$\left(\frac{\partial \log p_O(1)}{\partial \log Re}\right)_{Q,p_0(0)} > 0 \Rightarrow \text{INSTABILITY}$$



COMPLETE INSTABILITY MODEL

$$p(x,y,z,t) = p(x)\,e^{i(k_r r) + \omega t} \tag{33}$$

$$\frac{dp}{dx} = -\left(b_1 T_{g0}^{v} + 2b_2\right) u_x - \left(b_1 u_0 v T_{g0}^{v-1}\right) T_g - \left(b_2 u_0^2\right) \rho \tag{34}$$

$$\frac{dT_g}{dx} = H\left(T_p - T_g\right) - \left(\rho_0 \omega\right) T_g \tag{35}$$

$$\frac{d\rho}{dx} = \left(\frac{1}{T_{g0}}\right)\frac{dp}{dx} - \left(\frac{q}{T_{g0}^2}\right) p - \left(\frac{\rho_0}{T_{g0}}\right)\frac{dT_g}{dx} - \left(\frac{1}{T_{g0}}\frac{d\rho_0}{dx} - \frac{q\rho_0}{T_{g0}^2}\right) T_g \tag{36}$$

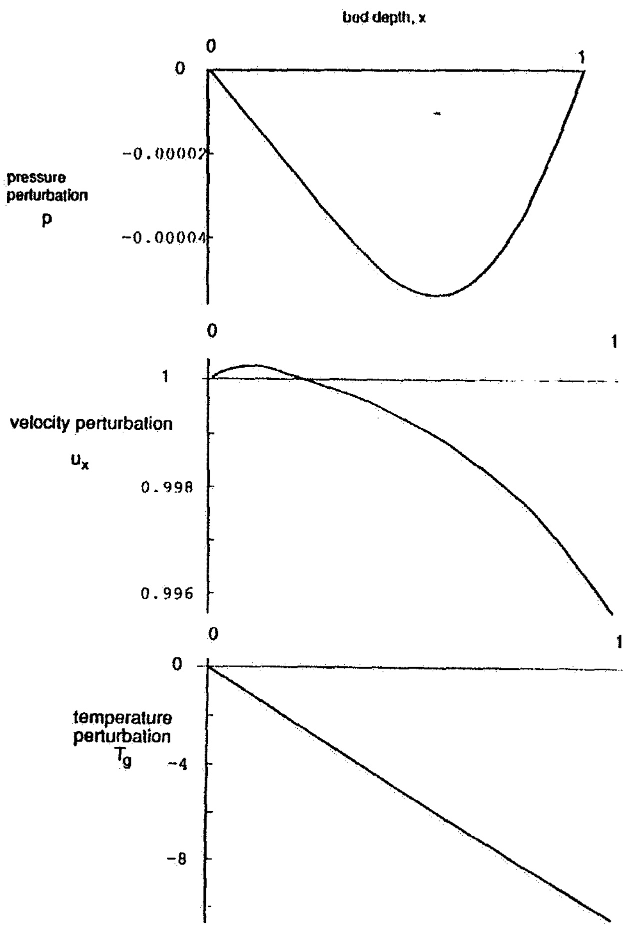
$$\frac{du_x}{dx} = -\left(\frac{1}{\rho_0}\frac{d\rho_0}{dx}\right) u_x - \left(\frac{u_0}{\rho_0}\right)\frac{d\rho}{dx} - \left(\frac{1}{\rho_0}\frac{du_0}{dx} + \frac{\omega}{\rho_0}\right)\rho - \left(\frac{k_r^2}{b_2 + b_1 T_{g0}^{v}}\right) p \tag{37}$$

$$K\frac{dy}{dx} = \left(Kk_r^2 + c\omega + H\right) T_p - H\,T_g + \left(q\rho_0\right) u_x + \left(q u_0\right)\rho \tag{38}$$

$$\frac{dT_p}{dx} = y \tag{39}$$

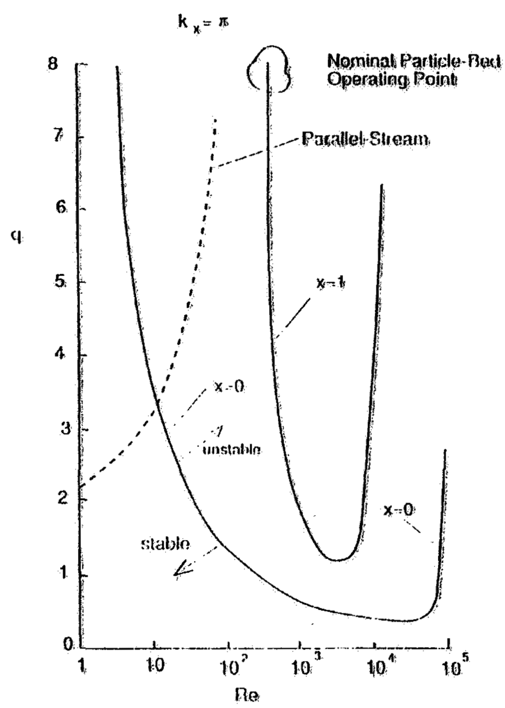
Approximate Case Neglecting Conduction in x:

$$T_p = \frac{\left(H\,T_g - (q\rho_0)\,u_x - (q u_0)\,\rho\right)}{\left(Kk_r^2 + c\omega + H\right)} \tag{40}$$

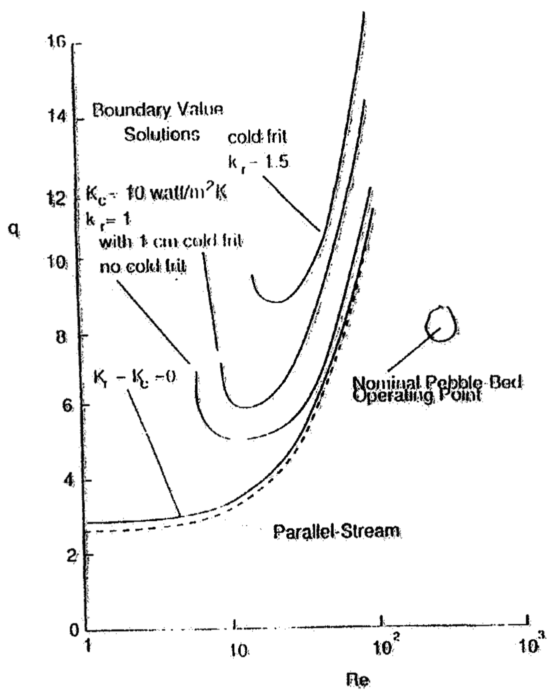


LOCAL INSTABILITY

$$p(x,y,z,t) = p(k_x,k_y,k_z,\omega)\, e^{i(\vec{k}\cdot\vec{x}) + \omega t}$$



COMPLETE INSTABILITY MODEL



N93-26935

NUCLEAR PROPULSION TECHNOLOGY

ADVANCED FUELS TECHNOLOGY

WALTER A. STARK, JR.
LOS ALAMOS NATIONAL LABORATORY

OCTOBER 21, 1992

NP-TIM-92

Los Alamos

NTP REACTOR & FUEL REQUIREMENTS

REACTOR REQUIREMENTS		FUEL REQUIREMENTS
PERFORMANCE:		
Specific Impulse	>925 sec	Fuel Temperature > 3000K
Thrust-to-Weight	>8	Uranium Loading > 0.8 g/cc
Single Burn Time	1 hr	Thermal & Chemical Stability
Operating Life Time	10 hr	Low Diffusion Rates
Restart s	>10	Thermal Shock Resistance
SAFETY:		
ALARA radiation		FP retention
Large margin to failure		High Melting Point
Redundancy		Robust Fuel Elements
Fast restart		Thermal Shock Resistance

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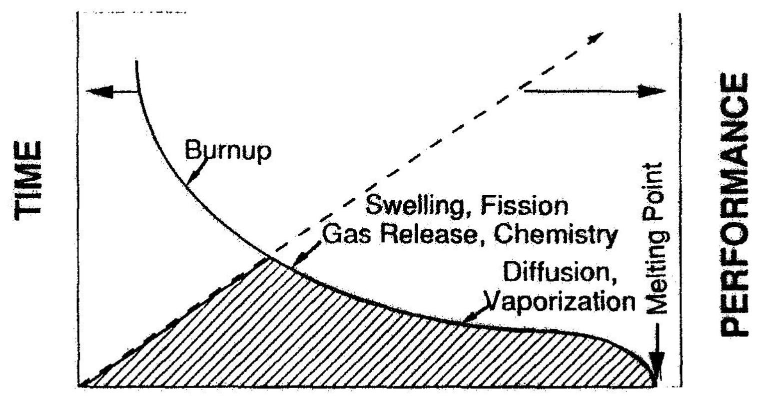
NUCLEAR PROPULSION TRADE-OFFS



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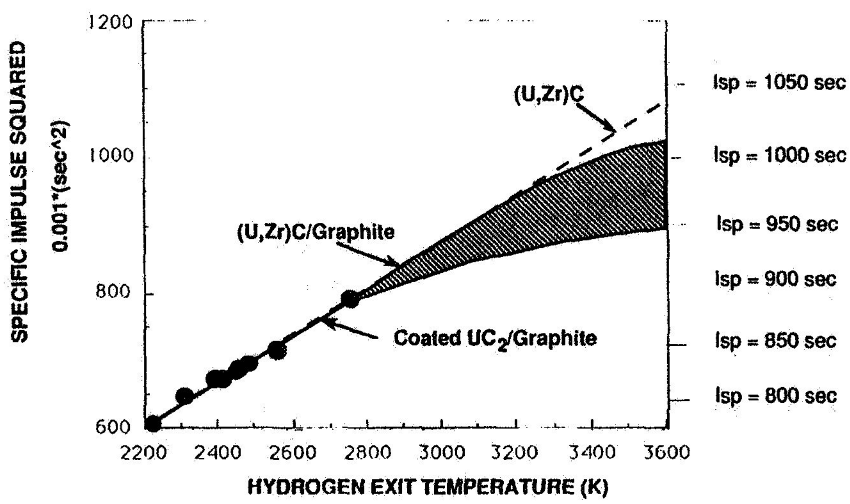
PROPULSION EFFICIENCY AND TEMPERATURE



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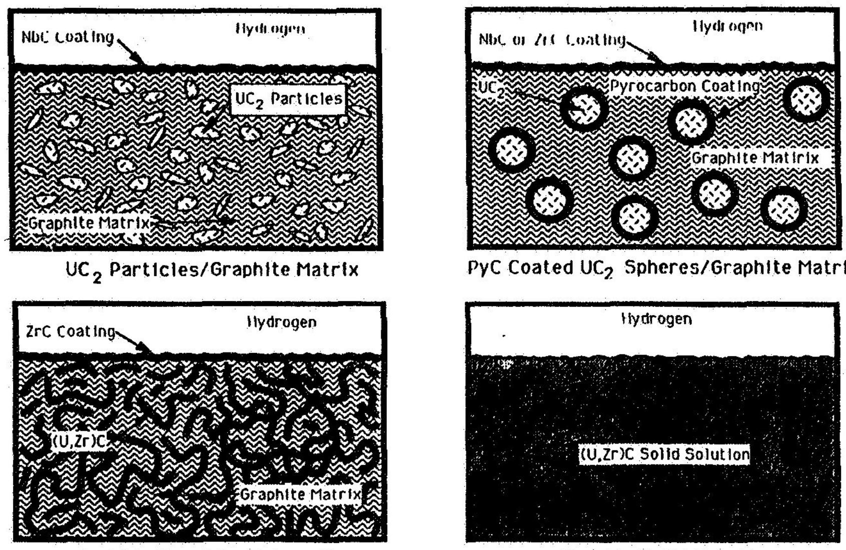
ROVER FUEL TYPES



UC_2 Particles/Graphite Matrix — PyC Coated UC_2 Spheres/Graphite Matrix

Carbide/Graphite Composite — Carbide Solid Solution

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URANIUM FUEL COMPOUNDS

Property	UO_2	UC	UC_2	UN	$U_{.2}Zr_{.8}C_{.99}$
Density, g/cc	10.96	13.63	11.68	14.32	8.01
U Density, g/cc	9.66	12.97	10.60	13.52	2.88
Melting Point, K	3100	2775	2710	3035*	3350
Thermal Expansion, 10^{-6} / K (@ 1273 K)	10.1	11.2	12.0	8.9	7.6
Thermal Conductivity, W/cm K (@ 1273K)	0.035	0.23	0.07	0.25	0.3

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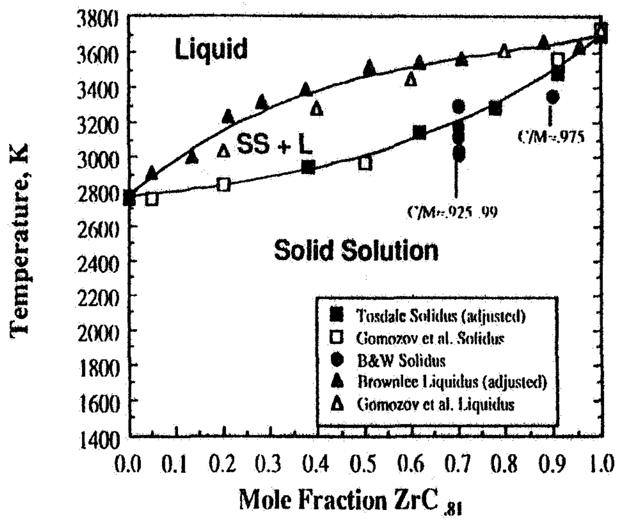
UC-ZrC PSEUDO-BINARY PHASE DIAGRAM



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MAJOR SOURCES OF DATA U-Zr-C PHASE DIAGRAM

Uranium - Carbon Binary
Uranium Carbide - Zirconium Carbide Pseudo-Binary
Uranium Dicarbide - Zirconium Carbide Pseudo-Binary
Zirconium - Carbon Binary
Calculations - Chang Formulation
- Butt and Wallace

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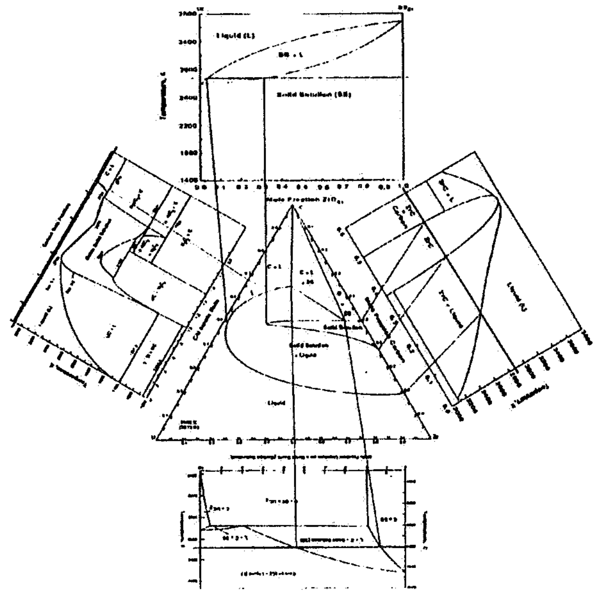
PHASE DIAGRAM "OPTIMIZATION"



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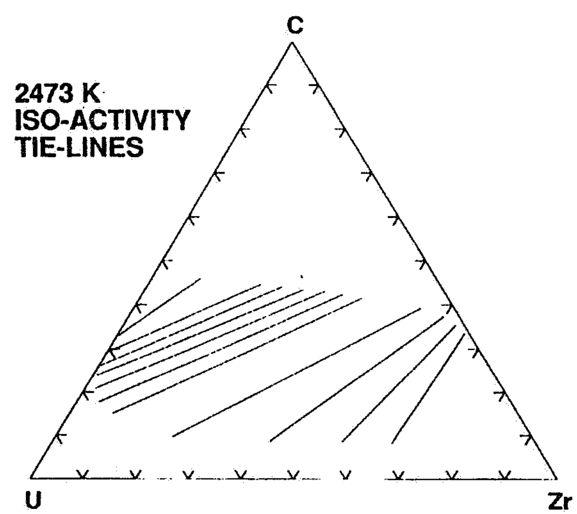
SOLIDUS - LIQUIDUS CALCULATION



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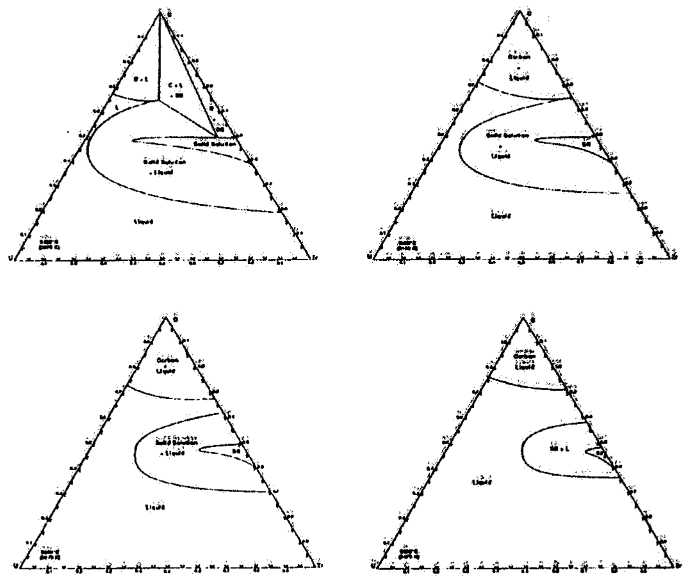
U-Zr-C ISOTHERMAL SECTIONS



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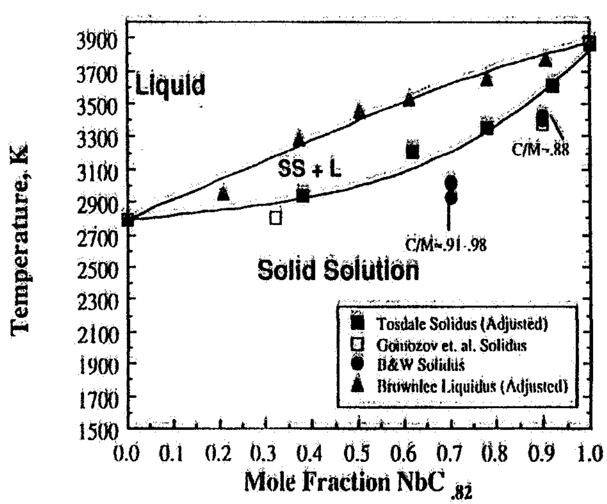
UC-NbC PSEUDO-BINARY PHASE DIAGRAM



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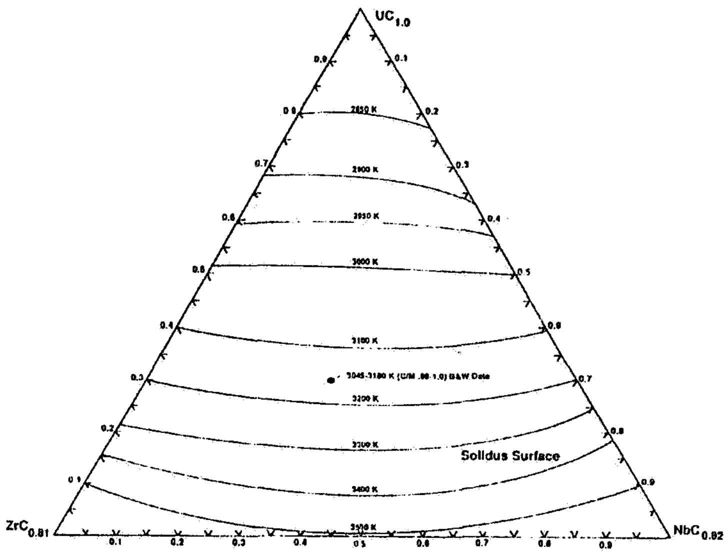
UC-NbC-ZrC PSEUDO TERNARY SYSTEM



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MELTING POINT EXPERIMENTS

SAMPLE FABRICATION

- COMPOSITION
- FABRICATION

MEASUREMENT

ANALYSIS

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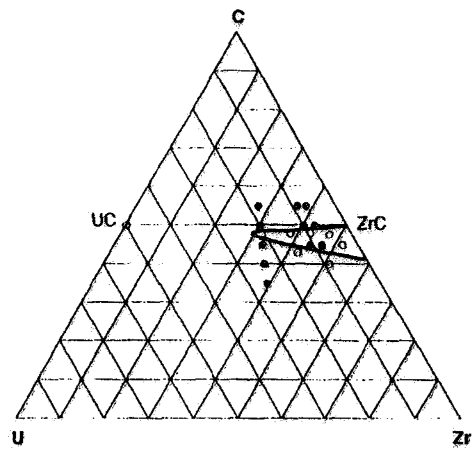
PRELIMINARY MELT POINT COMPOSITIONS



U-Zr-C Ternary Diagram
3273 K ISOTHERMAL SECTION

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SAMPLE FABRICATION TECHNIQUES

COLD PRESS, REDUCE, AND SINTER

ARC MELT

COMBUSTION SYNTHESIS

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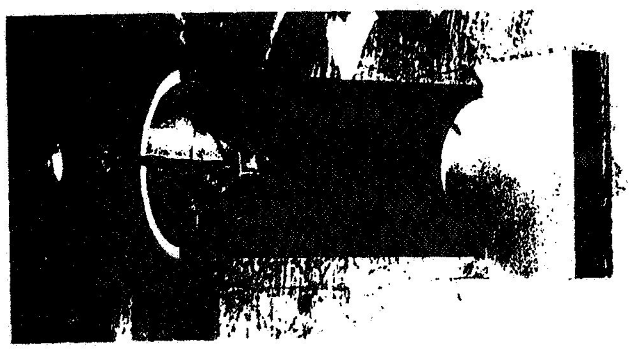
Meltpoint Setup



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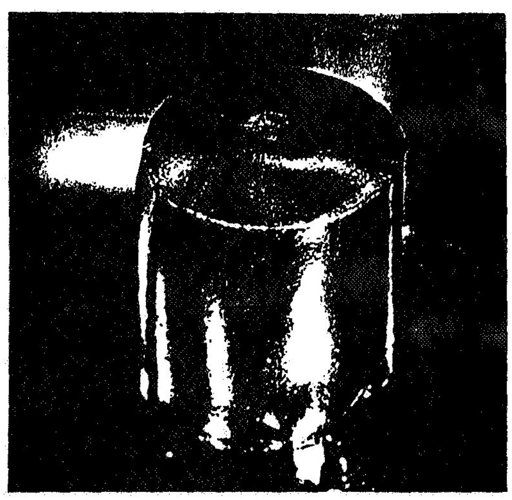
Post-Melt Sample



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MEASURED MELT POINT COMPARISON

Composition	Observed Melt Pt., K	Literature Value, K	Variance, K
$UC_{1.0}$	2806	2793	+13
$UC_{1.4}$	2633	2673	-40
$U_{.4}Zr_{.6}C_{1.2}$	2683	2673	+10
$U_{.4}Zr_{.6}C_{1.2}$	2655	2673	-18

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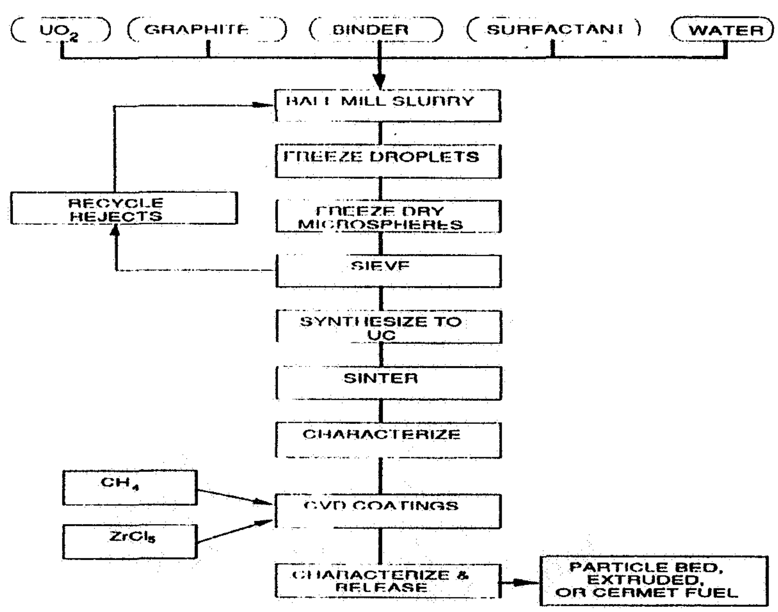
CRYOCHEMICAL FUEL PROCESSING



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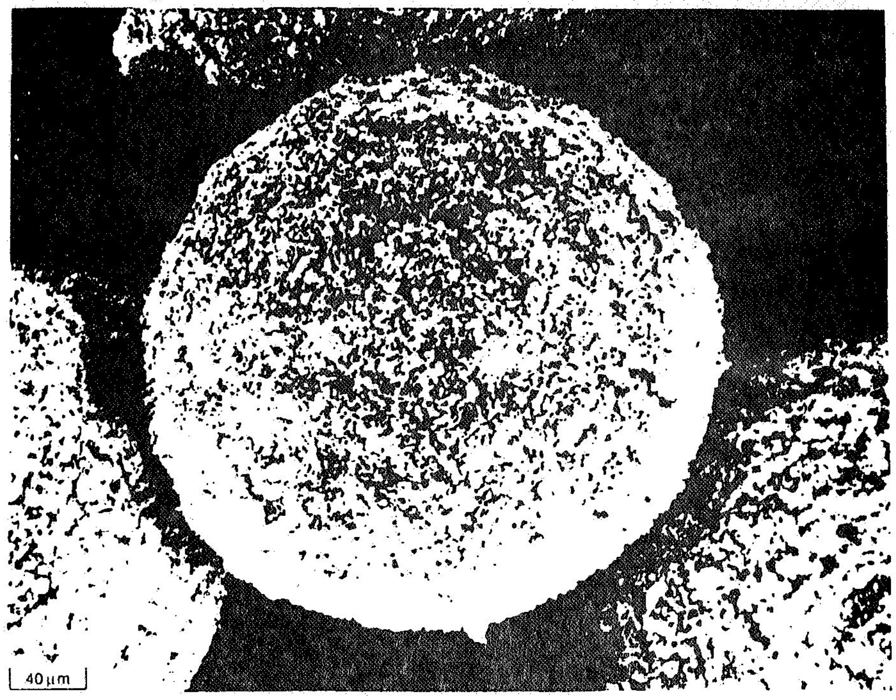
UO$_2$ + C MICROSPHERE AFTER FREEZE DRYING ≈500X



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SINTERED UC$_2$ MICROSPHERES

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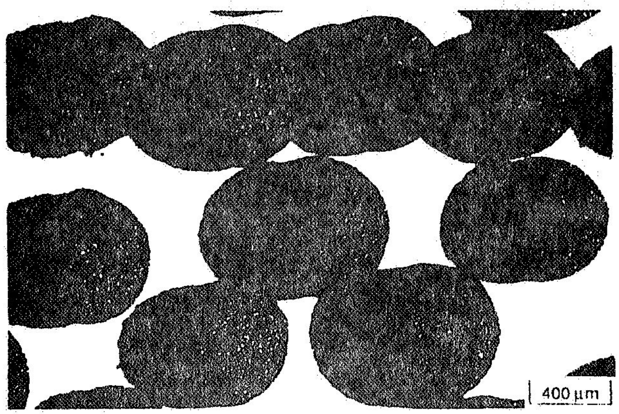
SINTERED $(U_{0.1}Zr_{0.9})C$ MICROSPHERES



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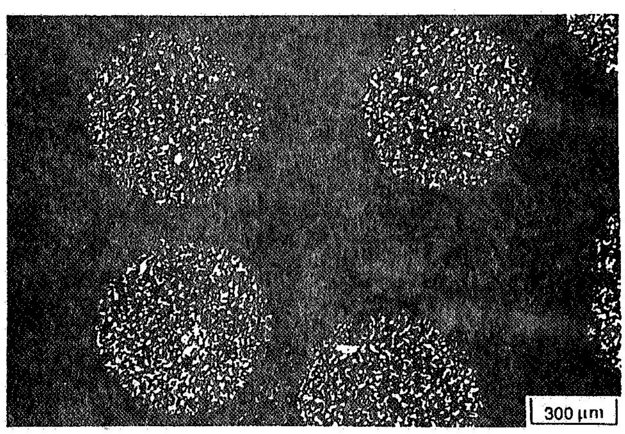
CROSS-SECTION OF SINTERED MIXED CARBIDE MICROSPHERES



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CRYOCHEMICAL SPHERE FORMING ADVANTAGES

- Process is composed of a few simple steps
- Applicable to a variety of nuclear fuel concepts
- Porosity is likely a controllable variable
- Spheres >1000 μm diameter appear possible
- Rejected spheres are easily reused
- Re-using process fluids minimizes wastes

Alamos

Laser Diagnostics for NTP Fuel Corrosion Studies

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Contributors: ***D.P. Butt***
A.D. Sappey

Nuclear Propulsion Technical Interchange Meeting
NASA-Lewis Research Center, Plum Brook Station

October 20-23, 1992

Issues Associated with NTP Fuels

- Reactor operates at temperatures approaching 3000 K
- Fuel is susceptible to corrosion
- Need to quantify corrosion products / rates to assess fuel performance
- Severe environment during test: limits methods of measurement

Laser-based diagnostics offer means of probing environment about fuel

Advantages: non-intrusive
unaffected by harsh local conditions

CLS-92-1582

In order to generate the necessary propulsive performance, the nuclear reactor will operate at temperatures approaching 3000 K. Such temperatures, in combination with the high-pressure hydrogen propellant flowing through the core, provide a very hostile operating environment for the fuel material, particularly with respect to hydrogen-induced corrosion. Identifying the corrosion products as well as experimentally quantifying the corrosion rates of these fuels under a variety of conditions is critical to assess the expected lifetime of the reactor. The severe environment surrounding a fuel material under test is not conducive to probing by standard instrumentation. Thus alternative diagnostic techniques are required to provide real-time monitoring of corrosion products. Laser diagnostics offer a means of non-intrusively probing the high-temperature environment above the surface of the fuel to identify and establish spatial distributions and local concentrations of many of the anticipated corrosion species.

U-Zr-C system corrosion products

- $(U_x Zr_{1-x}) C_y$ solid solution or $(U_x Zr_{1-x}) C_y + C$ composite fuel material serious candidate for NTP reactor fuels
  - High melting temperature
  - Good resistance to H_2 corrosion
- Calculations indicate that for H_2 temperatures >2800 K, acetylene, U(g), and Zr(g) are primary vapor constituents
- Zr selected as species to probe
  - Major corrosion product
  - Corrosion of $(U_x Zr_{1-x}) C_y$ may be rate limited by transport of Zr away from fuel surfaces
  - Obviates need for testing in H_2 environments or with uranium-containing samples

CLS-92-1957

Solid solution refractory carbide fuel materials, such as $U_xZr_{1-x}C_y$, are being seriously considered as candidate NTP reactor fuels because of their high melting point and resistance to corrosion by hydrogen.[1] Butt[2] has calculated the equilibrium partial pressures above the surface of $U_{0.05}Zr_{0.95}C_{1.07}$ during exposure to 1 atm of hydrogen over a temperature range of 2000 to 3200 K. The results show that in high temperature hydrogen, U(g), Zr(g), and various hydrocarbon species dominate the gas-phase products. Above approximately 2800K, acetylene and gas phase uranium and zirconium are predicted to be the primary vapor constituents. Confirmation of such predictions through experimental measurements is critical to the development of accurate corrosion kinetics models.

In the current study, Zr atoms were selected as the species to probe using laser techniques. This selection was influenced by several factors. First, the zirconium atom is predicted to be a major corrosion product. Second, the corrosion of $U_xZr_{1-x}C_y$ in hydrogen may be rate limited by the transport of gas phase Zr away form the surface.[2] Third, testing a laser diagnostic for zirconium obviates the need to perform experiments in a hydrogen environment or with uranium-containing fuel samples.

1. S. K. Bhattacharyya, R. H. Cooper, R. B. Matthews, C. S. Olsen, R. H. Titran, and C. E. Walter, "Fuels and Materials for Space Nuclear Propulsion," AIP Proceedings of the 9th Symposium on Space Nuclear Power Systems, Albuquerque, New Mexico, Vol. 2. pp 681-691 (1992).

2. D. P. Butt, "Corrosion of $U_xZr_{1-x}C_y$ Nuclear Fuel Materials in Hydrogen Gas at High Pressures and Temperatures," IAF Paper No. 92-0570, 43rd Congress of the International Astronautical Federation, Washington DC (1992).

Planar Laser-Induced Fluorescence (PLIF)

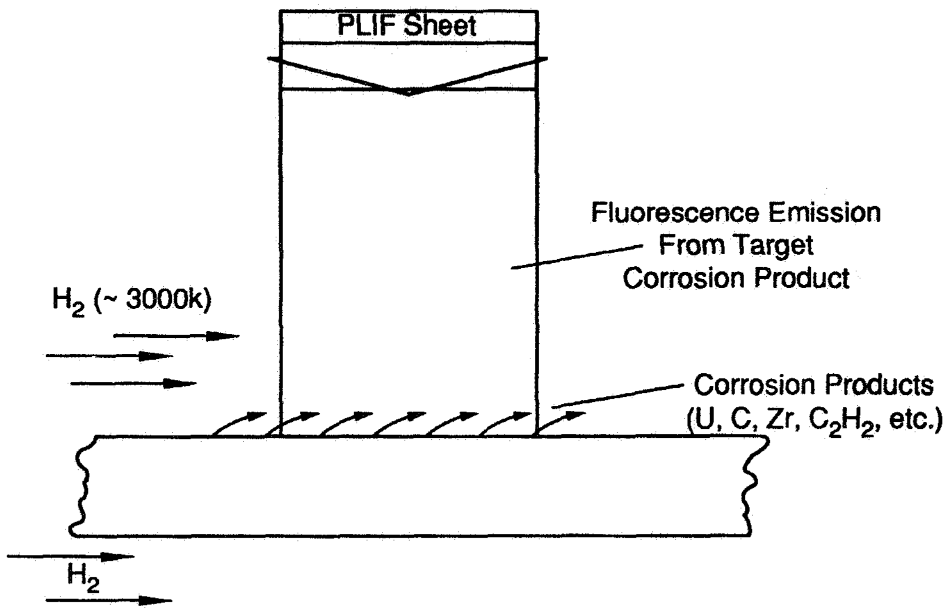
- Highly sensitive technique
- Absorption of laser light by atomic or molecular species followed after some finite time by emission from the excited state
- Intensity of emission can be related to concentration, temperature, velocity, etc.
- PLIF represents extension from point or line diagnostic to 2D or field measurement technique

CLS-92-1583

Laser-induced fluorescence (LIF) offers a highly sensitive technique for monitoring many of the NTP fuel corrosion products (including Zr) as well as for determining properties of the NTP exhaust. Quite simply, LIF can be viewed as an absorption of laser light, at a specific frequency, by an atomic or molecular species followed after some finite time by an emission from the excited state. This emission or fluorescence is, in general, at a different (typically longer) wavelength than the exciting laser light's wavelength. By viewing this off-resonance fluorescence, it is possible to avoid interference from scattered laser light.

Planar LIF or PLIF represents an extension of the LIF technique from a point or line diagnostic to a 2D or field measurement method. In general, the species-exciting laser beam is transformed into a thin sheet by the placement of cylindrical lenses into the optical train. A camera is typically used to collect the resulting fluorescence emission, perpendicular to the species-exciting laser sheet. The local intensity of the collected light can then be related to the concentration, temperature, or velocity of the target species. The non-intrusive nature of this technique, as well as its good spatial and temporal resolution, make it particularly well suited for application as a diagnostic in the high temperature NTP operating environment.

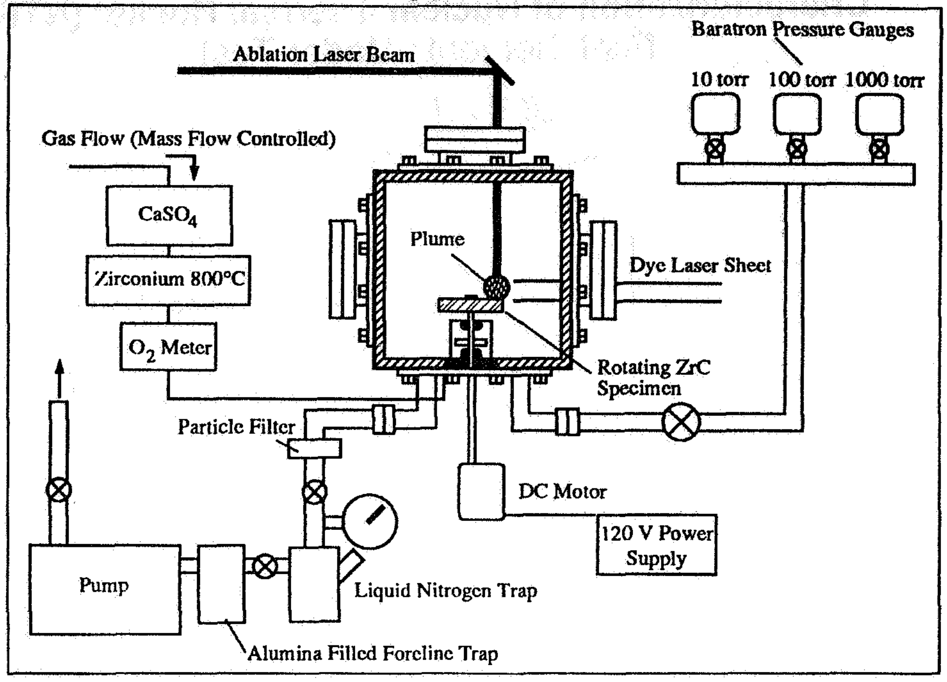
Utilization of PLIF for Corrosion Product Characterization of Nuclear Thermal Rocket (NTR) Fuel Elements Under Test



CLS-92-

A PLIF scheme for measurement of corrosion species evolving during the test of an NTP fuel sample/element could take the above configuration. The laser sheet is passed through the high temperature hydrogen stream, which contains the various corrosion products, and contacts the surface of the fuel. Fluorescence emission from the chosen target species is then imaged producing a two-dimensional distribution map.

Laser ablation / PLIF apparatus



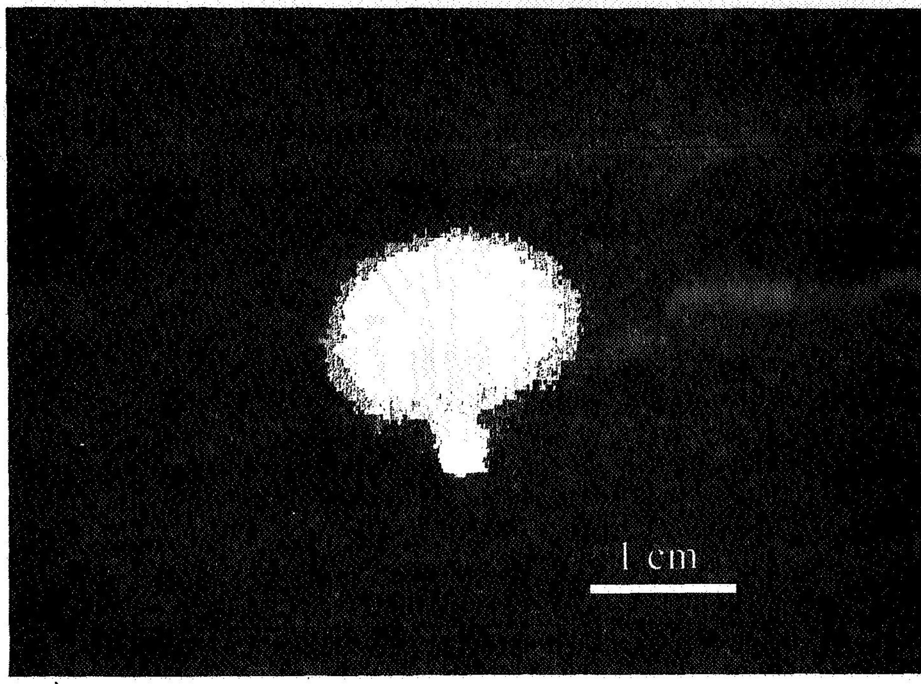
CLS-92-829

Production of zirconium vapor for subsequent illumination by a PLIF diagnostic, is accomplished by focusing a pulsed laser onto a ZrC target. This technique, known as laser ablation, represents a relatively simple method for producing gas phase samples of refractory materials.

The apparatus utilized for these experiments is displayed above. The cubical (~30 cm) ablation chamber contains five, window ports which allows optical access to its interior. The chamber is evacuated by a standard mechanical pump through both a liquid nitrogen and alumina-filled trap. The chamber can be backfilled with a variety of gases through a separate, flow regulated feed line. During each experiment, a slow flow of argon is maintained through this line and the chamber to minimize the buildup of particulate. A series of capacitance type manometers and thermocouple gauges are available to monitor chamber pressure. The base pressure for the chamber is 20 mtorr. Typical operating pressures are between 7 to 10 torr. The ZrC target specimen is positioned on a rotating table which is externally driven by a variable speed DC motor. Rotation of the target prevents the formation of a pit in the ZrC disk by action of the ablation laser.

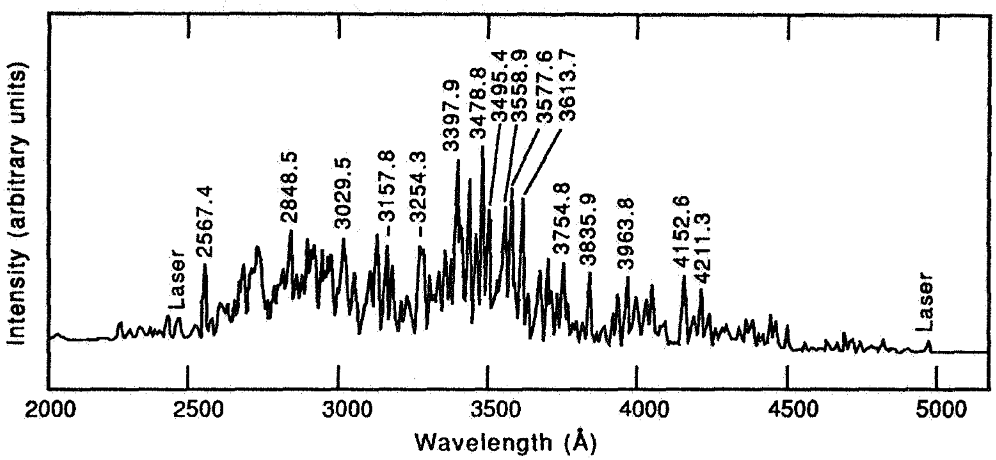
A Lumonics model TE-860-4 excimer laser operating at 248 nm is used to produce the abation pulse. Beam energies are on the order of 100 mJ/pulse and the laser is operated with a 5-10 Hz repetition frequency. The beam is brought to the ablation chamber by several high reflectance mirrors and focused at normal incidence onto the target using a 18.3-cm focal length quartz lens. The ablation spot size is approximately 1 mm^2 and the corresponding laser fluence at the target is ~670 MW/cm^2. The PLIF sheet is passed, at right angles, through the plume.

ZrC Plume Emission



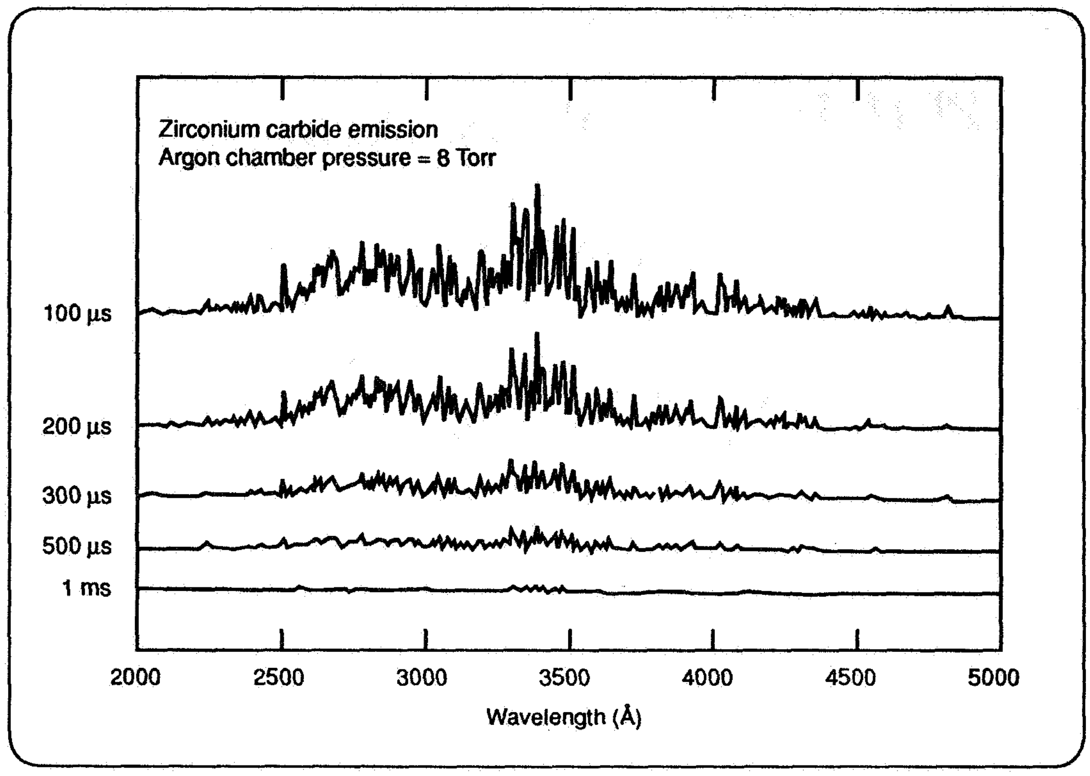
Plume emission images were recorded at different delay times. One representative ZrC plume emission image is displayed above. The ability to acquire such images represents a necessary step in the application of the PLIF technique. An intensified, gated uv camera (Xybion model ISG-250-U) with a 105 mm, f/4.5 quartz focusing lens was interfaced with an EPIX Silicon MUX RGB frame grabber board with a programmable trigger option. The timing of the camera intensifier, frame grabber board, and excimer laser was controlled using a Stanford Research System model DG535 programmable delay/pulse generator. The above image was captured 50 μs after the excimer laser pulse. The ablation chamber's argon background pressure was held constant at 7.5 torr and the camera gate width set at 20 μs. In these expanding plasmas, atom velocities can exceed 10^6 cm/s at low background pressures (tens of mtorr) with neutral gas temperatures near the target surface approaching 15,000 K.

ZrC Emission Spectrum



CLS-92-794

Temporally resolved ZrC plume emission spectra were recorded for regions near the target surface. The chamber pressure was maintained at 8 torr. A representative ZrC plume emission spectrum, recorded 10 μs after the excimer laser pulse, is shown above. This emission spectrum, which covers a wavelength range from 200.0 to 500.0 nm, is dominated by the presence of zirconium atom emission (several of the many Zr(I) lines are identified in the figure). No emission lines from other species, such as carbon atoms are identified in the emission traces.



CLS-92-1590

By adjusting the delay time, it is possible to establish emission spectra at specific times during the plume expansion event. For example, ZrC emission spectra recorded for increasing delay times indicate that the plume emission intensity has reduced to essentially undetectable levels at approximately 1 ms (for a background pressure of 8 torr). Such a reduction is due to expansion cooling and quenching through radiative and collisional processes. Quantifying the temporal behavior of plume emission intensity is important for establishing the proper delay times to acquire PLIF images of the expanding zirconium vapor as it reduces interference from background emission.

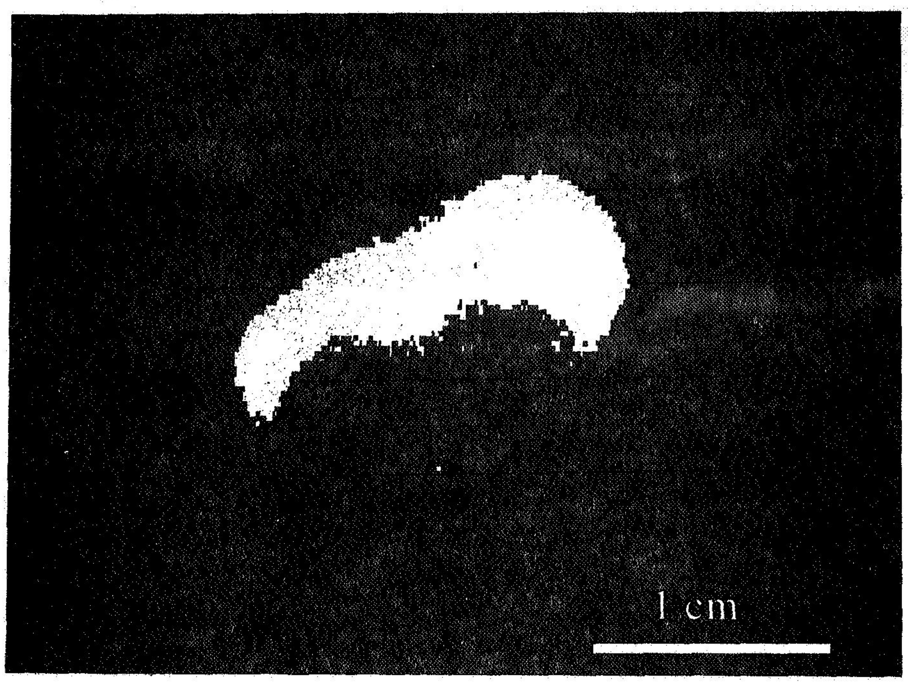
PLIF Imaging of ZrC Plume

- Zr(I) ground state to excited state transition at 35515 cm^{-1} pumped using Nd:YAG laser-pumped dye laser (R590 dye)
- Fluorescence emission monitored at 418.76 nm using filtered, gated, uv-intensified CCD camera coupled to frame-grabber board
- Image capture at different times after excimer pulse
- PLIF sheet 30.0 mm × <0.5 mm

CLS-92-1584

PLIF images of the spatial distribution of the zirconium atom in the ZrC plume have been acquired. To capture these images, the Zr(I) ground state to the excited state ($t^3F^o_2$, J=2) transition at 35,515.4 cm^{-1} (281.5 nm) was pumped using the frequency doubled output of a Nd:YAG laser-pumped dye laser operating on Rhodamine 590 dye. Fluorescence emission at 418.76 nm, corresponding to the transition from the $t^3F^o_2$ excited state to the b^3F_2 (J=2) intermediate state at 11640.7 cm^{-1}, was captured with the uv-intensified CCD camera. A dye laser sheet approximately 30.0 mm by 0.5 mm was formed for passage through the plume by a lens combination consisting of a +100 mm (converging) lens and a -100 mm (diverging) focal length cylindrical lens and a 150 mm focal length spherical lens.

PLIF Image of Zr in Ablated Plume



A PLIF image showing the spatial distribution of ground state zirconium atoms in the ZrC plume is shown above. The fluorescence emission was filtered using a GG395 filter. The image was recorded 625 μs after the ablation laser pulse with the camera gate width set at 20 μs. The spots observed in this image are pieces of hot (radiating) sputtered material from the target.

Summary

- **Utilized a focused excimer laser to ablate material from ZrC targets**
- **Zr prevalent in plumes**
- **Temporally resolved CCD image of plume emission generated**

$$\tau_{plume} \sim 1 \text{ ms } (P_{Background} = 8 \text{ torr}); \quad T_{exc} \cong 12{,}000 \text{ K}$$

- **PLIF utilized to successfully image Zr atom distribution in plume**
- **PLIF technique should be able to monitor Zr about fuel element under test**

CLS-92-1585

We have utilized a focused excimer laser to ablate material from ZrC targets for the purpose of developing appropriate laser-based diagnostics for gas-phase, corrosion products from hydrogen-exposed $U_xZr_{1-x}C_y$ fuel elements proposed for NTP application. Temporally and spatially resolved emission spectra from the produced vapor plumes show the dominating presence of zirconium atoms. Temporally and spatially resolved images of the ZrC plume emission have also been recorded. The PLIF technique has been successfully used to image Zr atom distributions in the ablated ZrC vapor plume and thus could potentially be utilized to monitor Zr about fuel elements under test.

Future Activities

- **Investigate other fluorescence excitation wavelengths for Zr**
- **Expose samples to rf-heated, hydrogen containing flow and probe flowfield around and downstream of sample with PLIF diagnostic**
- **Investigate other NTP fuel materials; (U-Nb-C) system**
- **Quantify Zr, Nb, etc. concentration in terms of fluorescence emission**

CLS-92-1586

Potential future activities include investigating other zirconium atom excitation wavelengths to ascertain the optimal transition for obtaining PLIF images in the ZrC ablation plumes. Following such determination, ZrC samples will be exposed to radio-frequency (rf) heated, hydrogen-containing flows and the PLIF diagnostic will be used to measure Zr atom distributions in the region surrounding the samples or in the nozzle exhaust flow.

For this study, a radio frequency discharge driven flow system will be used to produce a continuous, high temperature, chemically-clean gas stream. The unique feature of this system, which has been described by Wantuck,[1,2] is the use of an inductively coupled plasma tube as a high enthalpy gas source. A 50 kW rf generator is used to supply power to the tube where the the gas is heated to between 5000 to 10,000 K. The system configuration will allow two different modes of ZrC sample heating, namely, placement of the sample within the plasma tube for direct inductive/plasma heating or positioning downstream of the nozzle exit for heating by the gas stream. The first configuration approximates corrosion species distribution in an NTP exhaust flowfield. The second configuration best simulates propellant flow over a fuel element. In both cases, a dye laser beam, operating at the same wavelength employed for the plume PLIF illumination studies, will be used to probe the nozzle exit flowfield or the region surrounding the gas-stream heated ZrC sample.

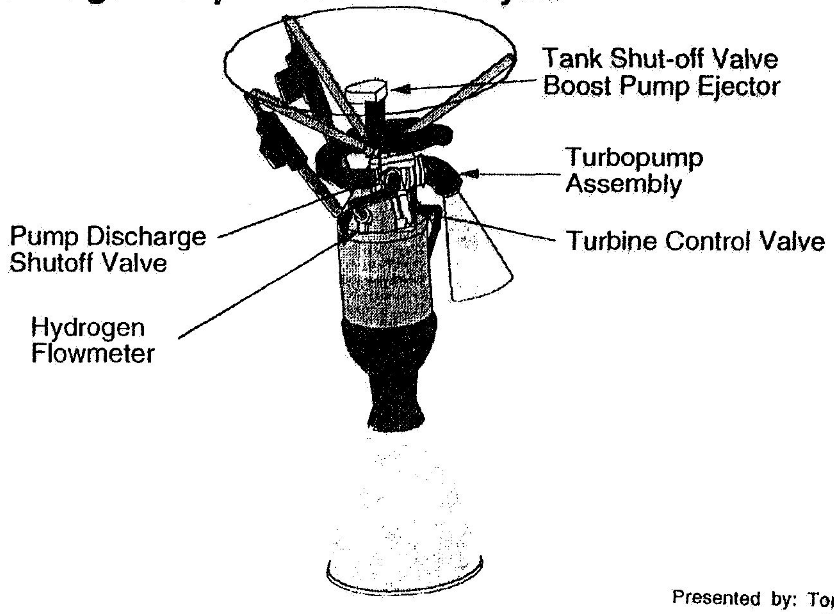
1. P. Wantuck and H. Watanabe, "Radio Frequency (RF) Heated Supersonic Flow Laboratory," AIAA Paper No. 90-2469, 1990.

2. P. J. Wantuck, R. A. Tennant, H. H. Watanabe, "Supersonic Combustion of a Transverse-Injected H_2 Jet in a Radio Frequency Heated Flow," AIAA Paper No. 91-2393, 1991.

N93-26937

SNTP Propellant Management System

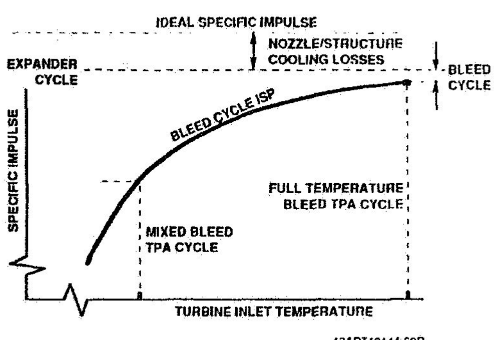
Current SNTP Engine System
Uses High Temperature Bleed Cycle



Presented by: Tom Tippetts
Allied Signal

SNTP Cycle Selection

Full-Temperature Bleed Cycle is Lowest Engine System Mass with Minimal Isp Penalty



- **No design interaction with reactor**
- **Allows light-weight radiation-cooled nozzle**
- **Lowest system complexity, potentially highest system reliability**
- **High-temperature, low-Z material minimize cooling in radiation environments**

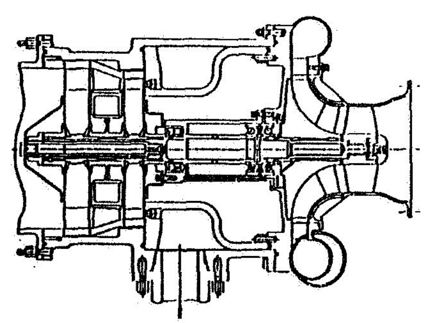
NTP System Components Have Unique Design Constraints

- High Ionizing Radiation Environment
- High Heat Load From Radiation Energy Absorption
- Restricts Use Of High-Z Materials
- Design Must Provide For Heat Removal

Bleed Cycle Presents Unique Design Requirements for Turbopump

SPACE NUCLEAR THERMAL PROPULSION

SNTP



- Moderate operating pressures (1350 psi)
  - Single-stage pump
  - Light pressure vessels
- High operating temperatures (2750 K)
  - Highly energetic working fluid
  - High-pressure ratio impulse turbine
  - High turbine temperatures
  - Large thermal gradients
- Environmental factors
  - Environmental heating — low -Z material
  - Limited elastomers selection
  - Hot-hydrogen embrittlement
- Use of bleed cycle and uncooled thrust nozzle results in substantial system weight savings.

Bleed-Cycle Turbopump

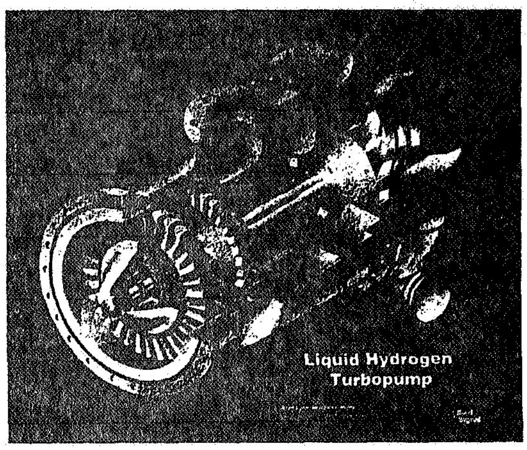
Uses Carbon-Carbon Components for Operation on 2750 K Gas

SPACE NUCLEAR THERMAL PROPULSION SNTP

Carbon-Carbon Hot Section Housing

Carbon-Carbon Turbines

Titanium Shafting



Carbon-Carbon Nozzle/Plenum

Aluminum Pump and Inducer

Ceramic Rolling Element Bearings or Foil Bearings

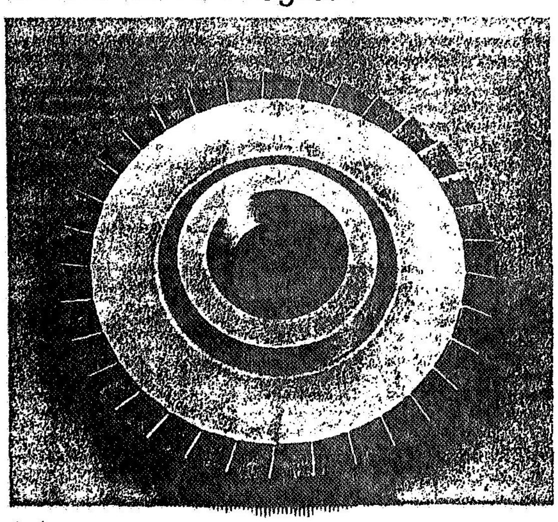
Page 4

(1)2782

SNTP Carbon-Carbon Turbine Wheel

Design is Based on Technology Developed on the ELITE Program

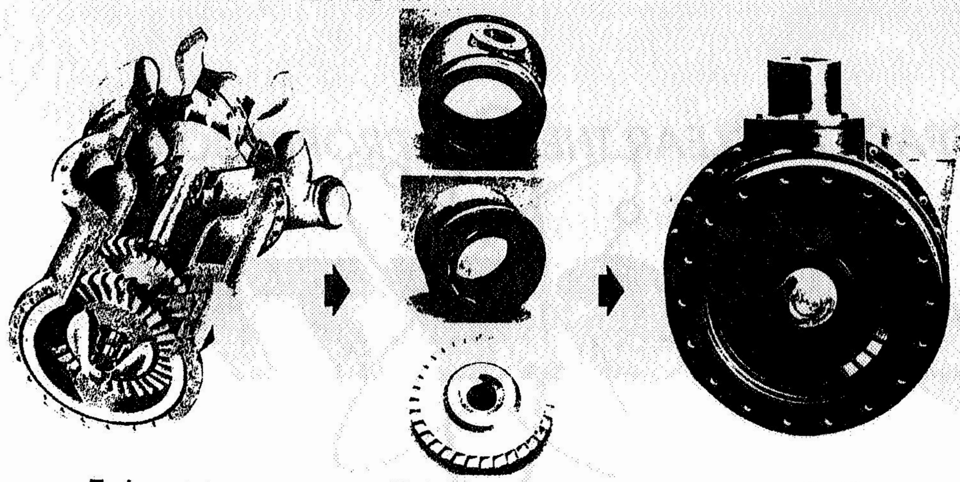
SPACE NUCLEAR THERMAL PROPULSION SNTP



- Helical 2-D polar weave architecture
- Impulse blades
- 55,600 rpm
- 2750 K inlet temperature
- 45-percent design stress margin
- 26-percent design speed margin

Turbine Development Program

High-Temperature, Carbon-Carbon Components Are Being Fabricated and Will be Tested at 2750 K



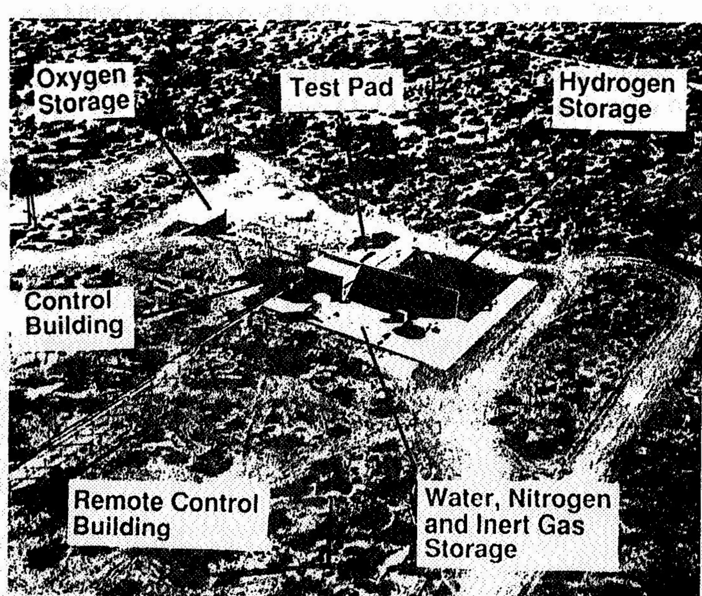
Turbopump Preliminary Design → High-Temperature Component Design and Fabrication → Hot-Turbine Spin Rig Test (HTSR)

Page 3

(T)2782

San Tan Hydrogen Test Facility

Facility Constructed for Development of SNTP Hydrogen-Related Components



- **Turbopump, valves, internal reactor components**
- **Hot, two-phase, and cryogenic hydrogen capability**
- **Dedicated facility for non-nuclear NTP testing**
- **Company-funded construction**

Page 1

N93-26938

SPACE NUCLEAR THERMAL PROPULSION

SNTP

US AIR FORCE
Phillips Laboratory
Grumman, Babcock & Wilcox
Brookhaven & Sandia National Labs

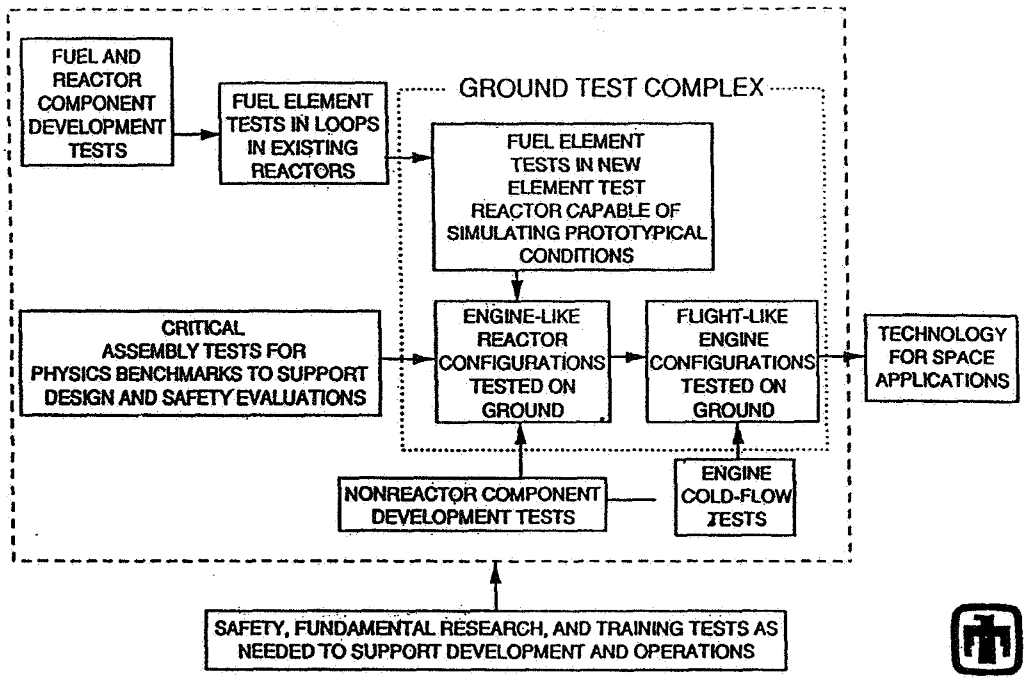
TESTS

PRESENTATION AT NUCLEAR PROPULSION
TECHNICAL INTERCHANGE MEETING

BY GEORGE C. ALLEN
SANDIA NATIONAL LABORATORIES

The objective of the SNTP program is to develop advanced nuclear thermal propulsion technology based on the particle bed reactor concept. A strong philosophical commitment exists in the industry/national laboratory team direced by the Air Force Phillips Laboratory to emphasize testing in development activities. This presentation focuses on nuclear testing currently underway to support development of SNTP technology.

Summary Test Logic For NTP Development



This is the summary test logic for NTP Development that has been generally accepted in the propulsion community provided resources are available. It is very consistent with the SNTP approach. Because of the limited time for the presentation, I will concentrate on critical assembly tests, fuel nuclear tests, and fuel element tests in loops in an existing reactor. Given time, I would discuss each box.

2

Critical Experiment (CX)

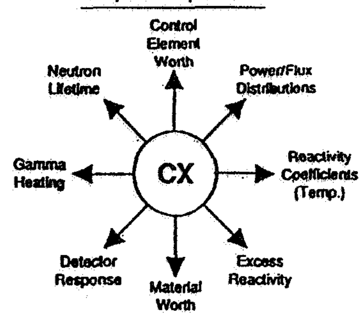
Purpose:

To obtain neutronic information for engineering needs (Safety, Mechanical Design, Control, Power Distribution, Weight, Reactor Design)

Accomplishments:

- Movable test reactor built - critical on October 24, 1989
- Excess reactivity - good agreement with predictions
- Peek-A-Boo control scheme feasibility demonstrated
- Control, safety, and shim element worth determined
- Moderator temperature coefficient determined
- Near-Term Priority:
  - Hot RHO experiment completion
  - Cold moderator follow - on experiment
  - Para/Ortho hydrogen worth

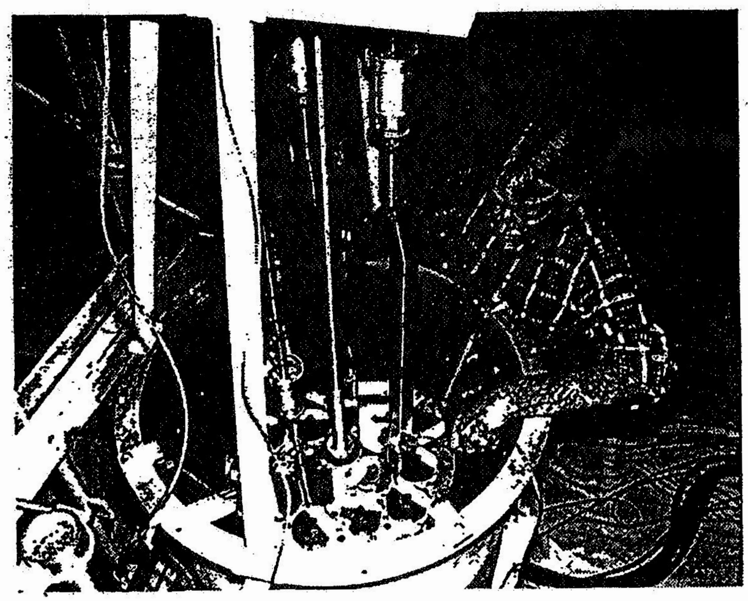
Completed Experiments



In the critical assembly test category, an operating low-power reactor called CX was developed and is operated in the Sandia Pulsed Reactor (SPR) facility. The goal of CX is to perform nuclear measurements that allow for neutronic codes and cross-sections to be benchmarked so that design margins, controllability, and limiting accident scenarios may be predicted and assessed for PBRs with reliability and certainty. The SNTP team successfully fulfilled all approval requirements imposed by DOE for operating a critical assembly. The CX achieved first critical in October 1989. Eight experiment campaigns have been performed to date with over 100 operations logged. Several of the major completed experiments have been listed in this vugraph.

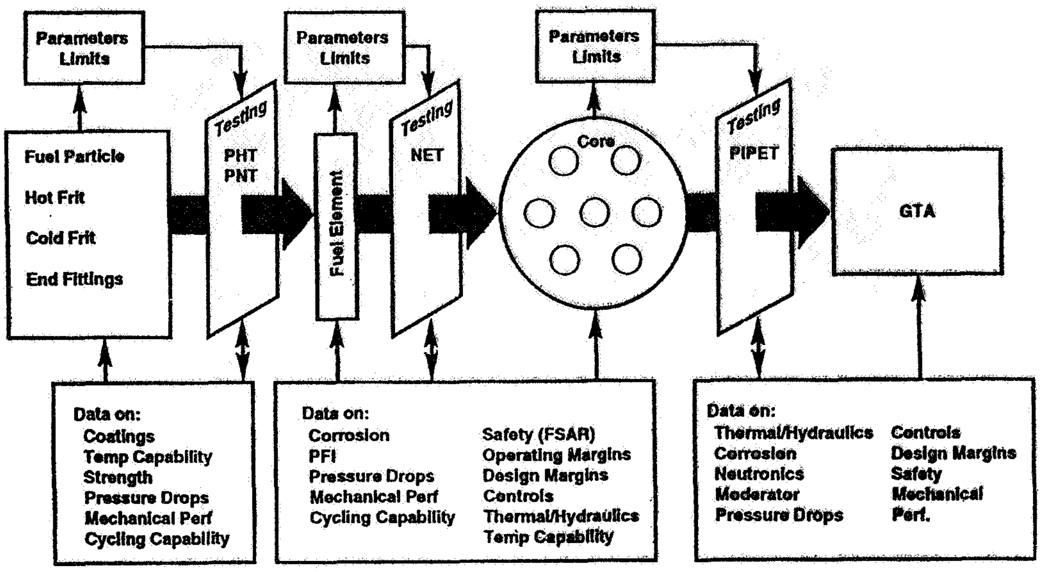
3

Sandia and the SNTP Team Built and Operate a Low-Power PBR for Reactor Physics Experiments



This is a photograph of the CX assembly with the water moderated removed.

Experiment Data Flow



Moving on to fuel and fuel element tests, this figure shows the experiment data flow for tests on these key nuclear components. Some of the test acronyms are as follows:

PHT = Particle Heating Tests
PNT = Particle Nuclear Tests
NET = Nuclear Element Tests
PIPET = PBR Integral Performance Element Tester
GTA = Ground Test Article (System-level test)

This sequence follows the test logic of going from components to subsystems to systems. The stepwise data achieved in these tests is key technology development and validation.

Particle Nuclear Test (PNT)

PURPOSE:

To conduct in-reactor testing of fuel particles to provide design verification, identify potential failure modes, and evaluate effects of manufacturing process variables.

ACCOMPLISHMENTS:

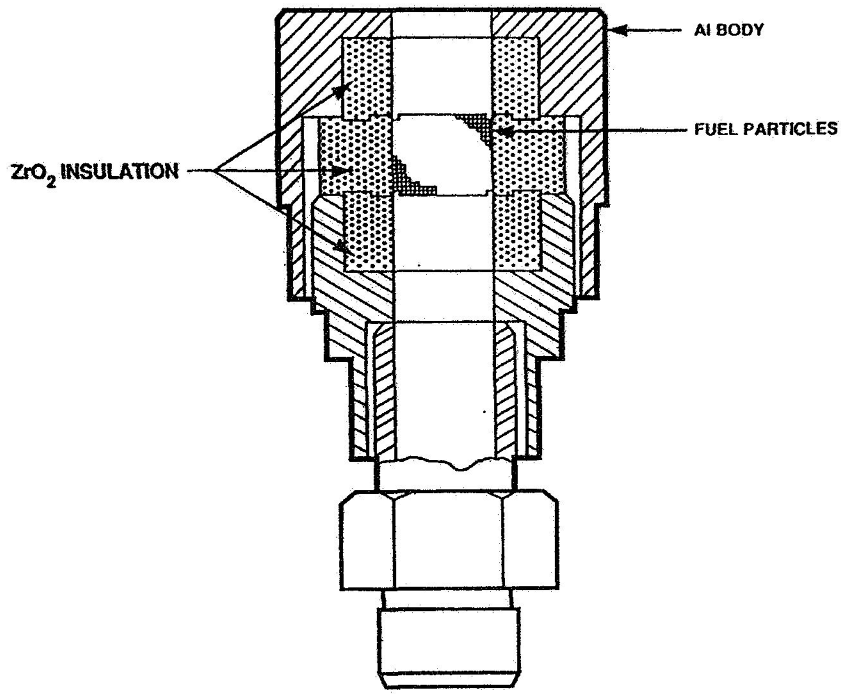
- 4 Capsule Tests
- 12 Fuel Holders
- 200,000 Fuel Particles
- 1800 - 3100K
- 150 - 600 Seconds
- 1 - 4 Cycles
- Fission Product Release Measurements

STATUS:

- **Performance limits determined for baseline fuel. Integrated fission product release Model operational. Additional tests planned as fuel becomes available.**

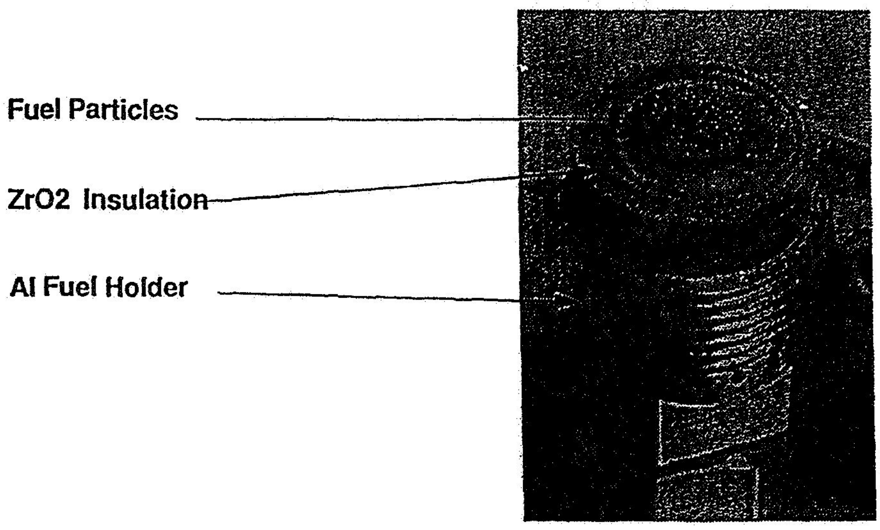
PNTs are performed on fuel particles to provide design verification, identify potential failure modes, and evaluate effects of manufacturing process variables. Since this vugraph was prepared, a fifth capsule test was performed on baseline particle fuels. This vugraph shows the range of test parameters. A computer code called HEISHI is now operational at Sandia to predict particle performance and to estimate fission product releases. The PNTs provide a considerable amount of data for code validation.

PNT-I Fuel Holder



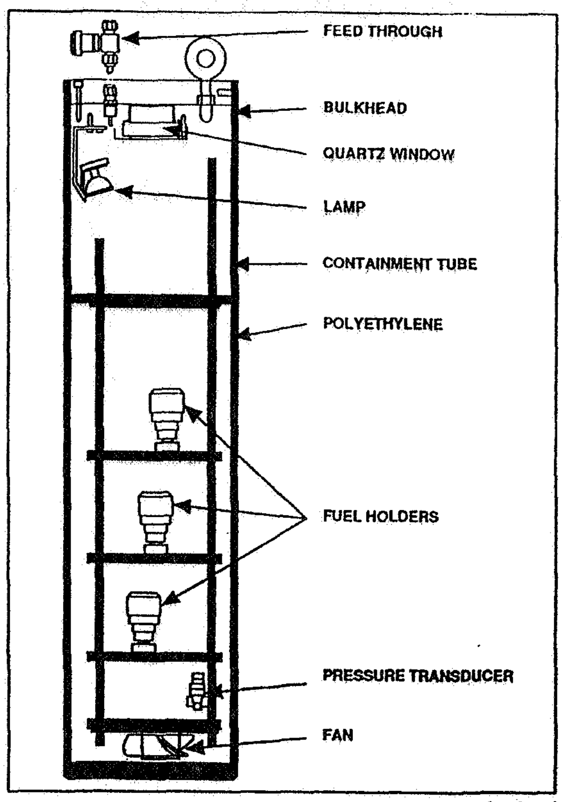
This is a fuel holder used in PNT experiments conducted to date. It holds one cubic centimeter of fuel particles.

PNT Fuel Holder



This is a photo of a PNT fuel holder with the top cap removed after a test.

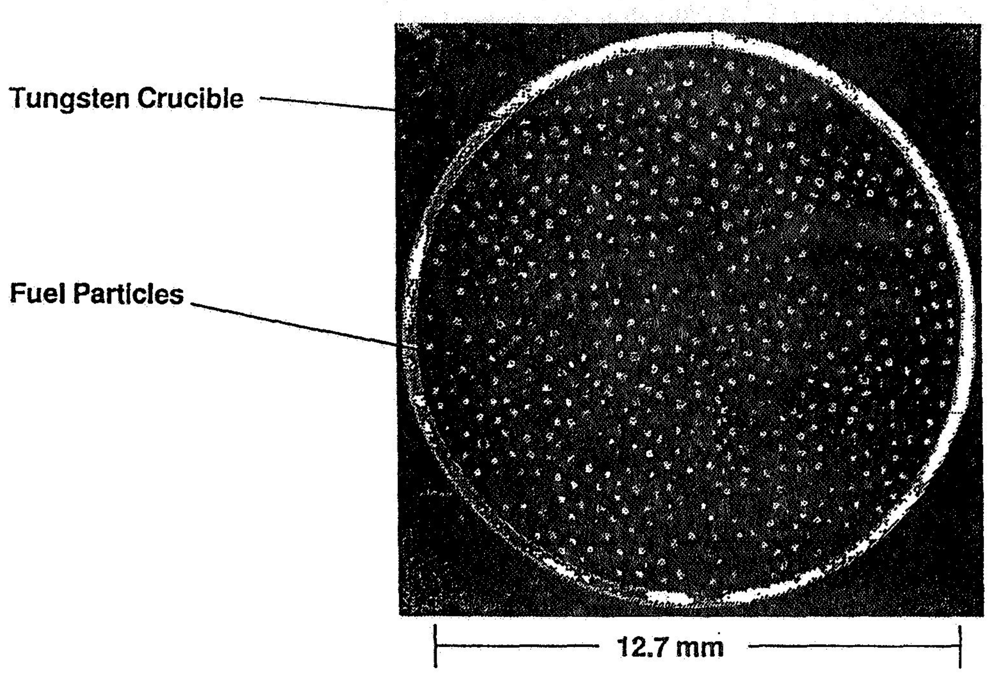
PNT-I Experiment Capsule



Several (typical 3) fuel holders are placed in an experiment capsule for irradiation. The capsule is placed in the central cavity of the Annular Core Research Reactor (ACRR). The ACRR is a pulse-type reactor that serves as the driver core for creating the desired experimental conditions. After an experiment is complete, the fuel is removed for post-irradiation examination.

9

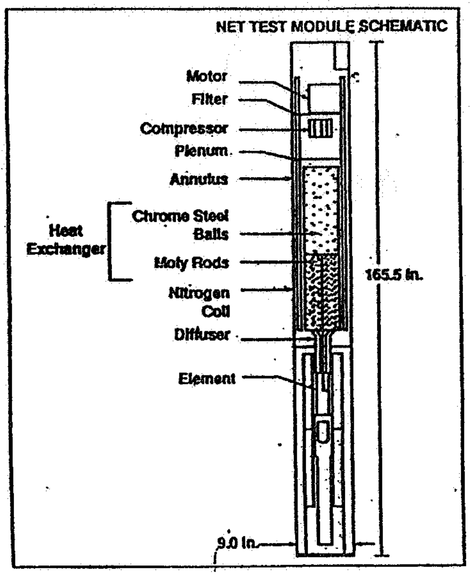
PNT Particle Bed After Nuclear Irradiation in the ACRR



This photo shows a PNT particle bed after nuclear irradiation in the ACRR.

Nuclear Element Test (NET)

- Closed Loop, In-Reactor Test Of A Complete Fuel Element In Flowing Cryogenic Hydrogen



Purpose:
- Demonstrate integration of fuel element technologies
- Test to full temperature capability
- Validate fuel element designs
- Support
  - -PIPET/GTA development
  - -Fuel development
  - -Model varification
  - -Safety analysis

Accomplishments:
- Test hardware designed and fabricated
- Unfueled experiment assembly and flow loop performance characterized in helium
- Test reactor (ACRR) control capabilities demonstrated

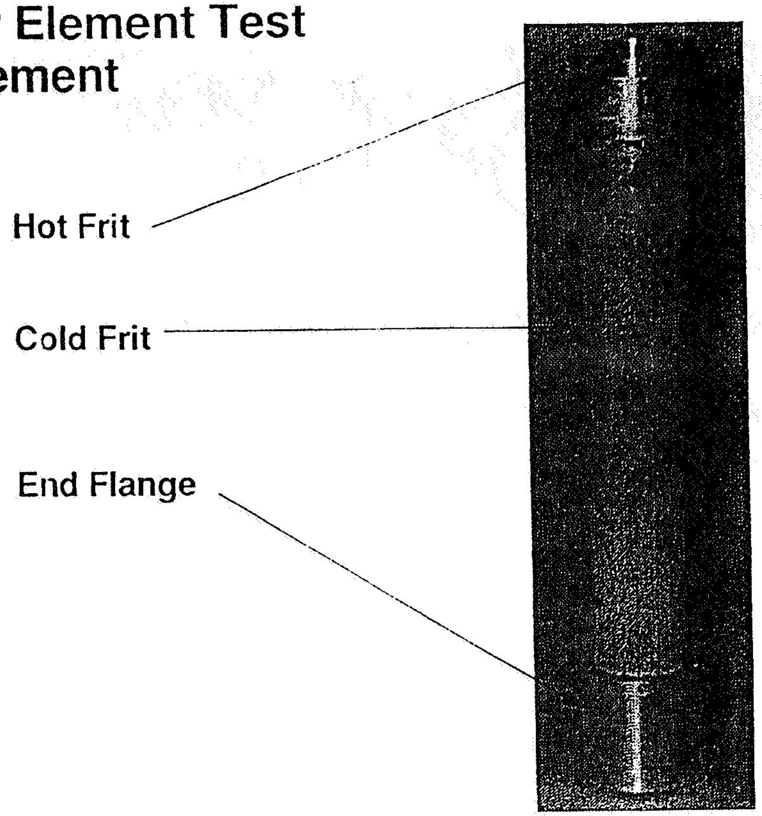
Status:
- Net-0 testing in cryogenic hydrogen summer 1992
- First fueled test (NET-1) early CY 1993

The next step is to assemble particles into a complete fuel element that can be tested incore in flowing hydrogen. The NET experiments provide this demonstration of integrated fuel element performance up to the limits of what environments can be achieved in existing reactor facilities. Since this vugraph was prepared, the unirradiated tests with cryogenic hydrogen in the NET-0 experiment capsule have been successfully completed. Six weeks after receipt of a fuel element, a nuclear test can be completed.

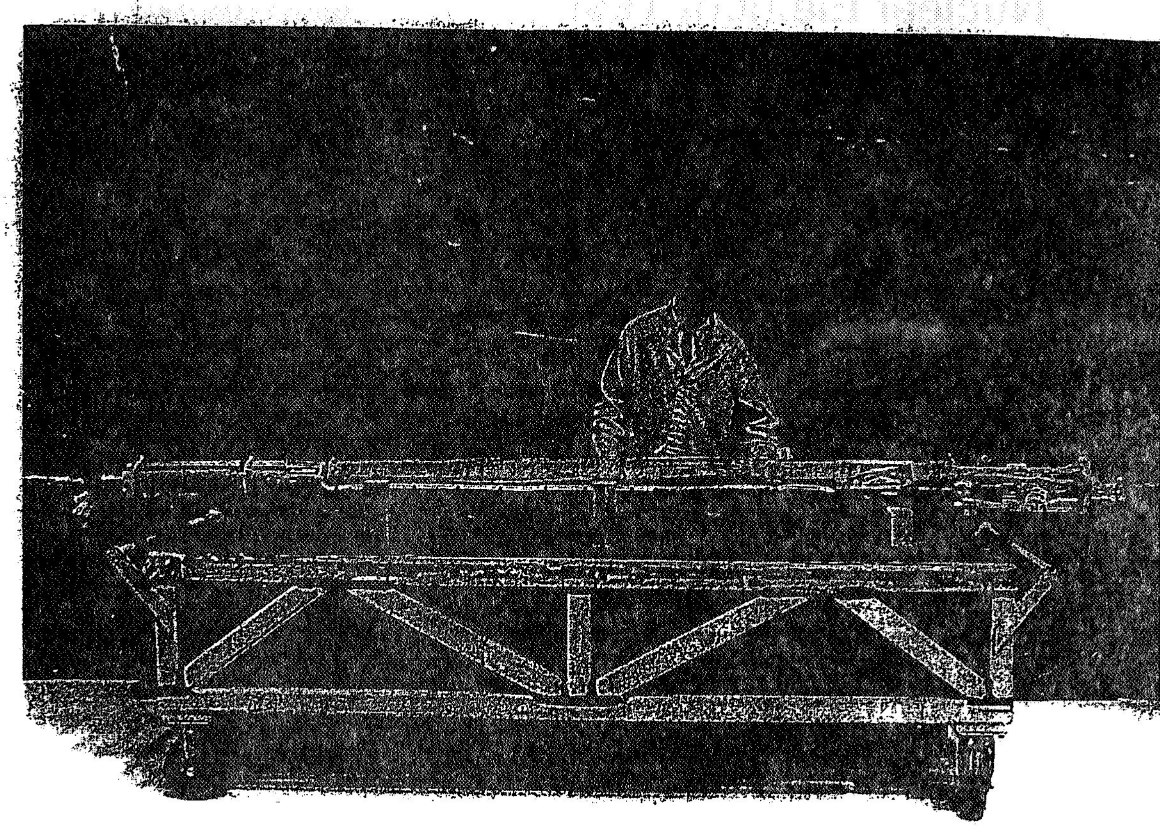
A computer code, F2D, has been developed and is used to predict the thermal-hydraulic performance of the NET element. Codes such as this are being used by the SNTP program to evaluate key thermal-hydraulic issues.

At the systems level, the SAFSIM Code has been developed. A separate paper on SAFSIM is being presented in a later session.

Nuclear Element Test
Fuel Element

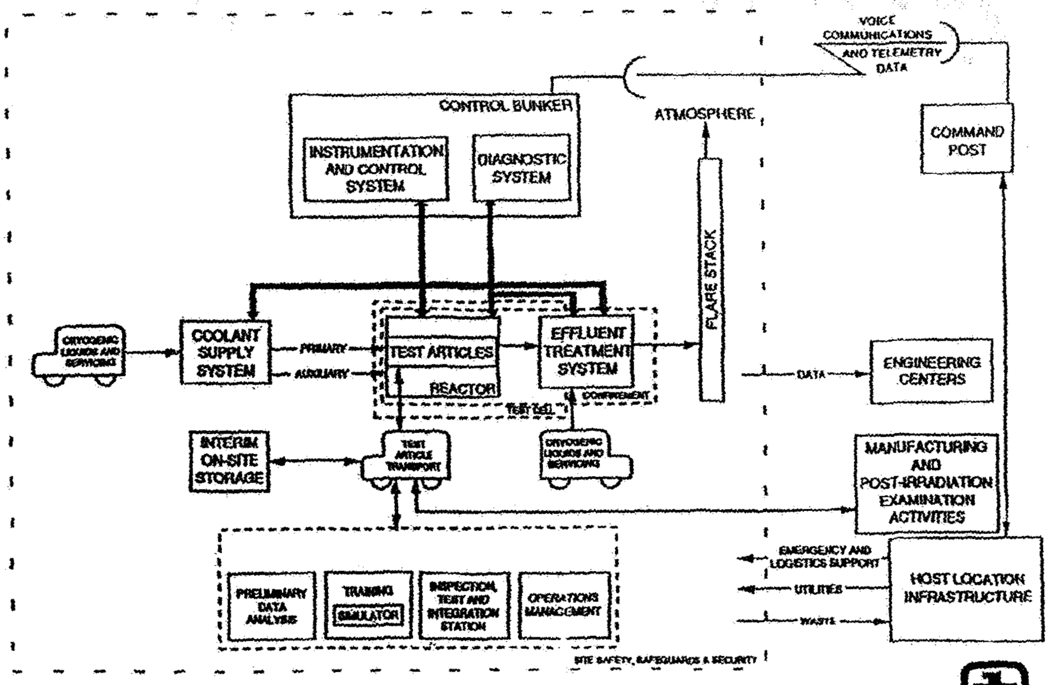


This photo shows the fuel element configuration to be evaluated in the first NET test.



This photo shows the inside portion of a NET capsule. It would be surrounded by the capsule containments for in-core tests.

Ground Test Facility System



In closing, I would like to note that extensive testing has already been performed and more will be conducted in the future using existing reactor facilities. However, eventually we will reach the limits of what we can do in existing facilities. A new ground test facility will be required for testing elements and reactor/engine systems. This vugraph shows the functions that must be performed by this facility. The environmental process to be reviewed by the next speaker outlines a key contribution to the decisions defining the scope and location of this ground test facility.

15

N93-26939

SPACE NUCLEAR THERMAL PROPULSION

SNTP

US AIR FORCE

PHILLIPS LABORATORY

SNTP ENVIRONMENTAL, SAFETY, AND HEALTH

NASA-LEWIS NP-TIM-92

Presenter

Charles D. Harmon

21 October 1992

SPACE NUCLEAR THERMAL PROPULSION

SNTP

US AIR FORCE

Phillips Laboratory

SAFETY POLICY

Implementation Plan and Goals

PROGRAM SAFETY POLICY

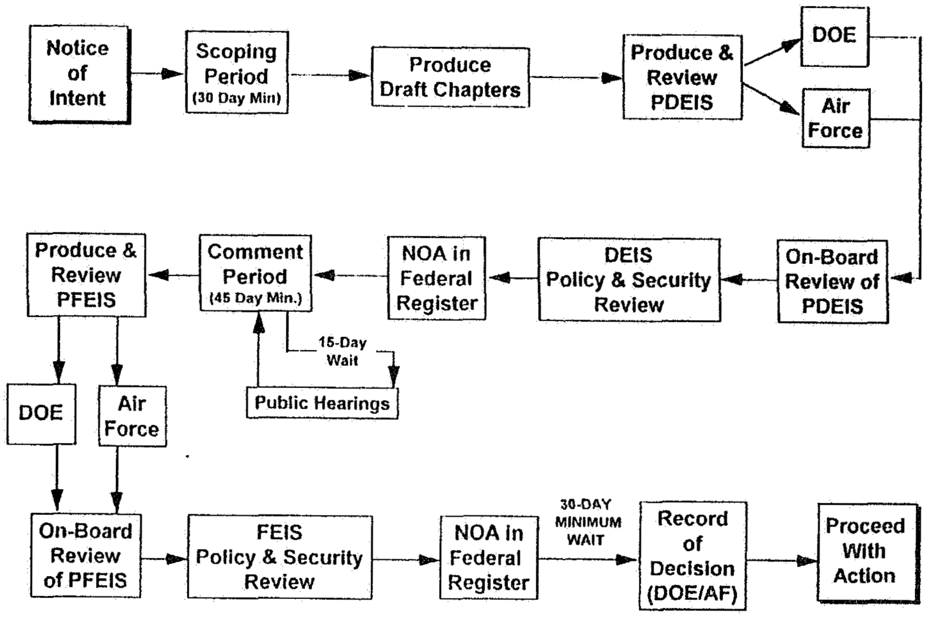
- o POLICY DOCUMENT PUBLISHED IN MAY 1992
- o OVERALL OBJECTIVES:
  - -- TO ENSURE THE MAXIMUM PROTECTION OF THE HEALTH AND SAFETY OF THE PUBLIC AND SNTP WORKERS
  - -- TO PROTECT PROPERTY FROM DAMAGE OR LOSS
  - -- TO PROTECT THE ENVIRONMENT FROM CONTAMINATION OR DAMAGE AS A CONSEQUENCE OF SNTP ACTIVITIES

PROGRAM SAFETY POLICIES

- o SAFETY AND ENVIRONMENTAL PROTECTION WILL BE EXPLICITLY CONSIDERED AND INCORPORATED THROUGHOUT THE LIFETIME OF EVERY SNTP PROGRAM ACTIVITY.
- o THE SNTP PROGRAM SHALL MEET ALL MANDATED, STATUTORY, AND LEGAL REQUIREMENTS FOR SAETY AND ENVIRONMENTAL PROTECTION.
- o ADDITIONALLY, EVERY PRACTICAL EFFORT SHALL BE MADE TO MAINTAIN RISKS DUE TO RADIATION AND TOXIC MATERIAL EXPOSURES AS LOW AS REASONABLY ACHIEVABLE (ALARA)
- o COMPLIANCE WITH THESE REQUIREMENTS WILL BE BASED ON THE PRINCIPLES OF DEFENSE-IN-DEPTH INVOLVING MULTIPLE PHYSICAL, PROCEDURAL, AND ADMINISTRATIVE BARRIERS.

Space Nuclear Thermal Propulsion

EIS Process



DEIS PUBLIC HEARING COMMENTS

- o NEPA PROCESS:
  - -- INSUFFICIENT REVIEW PERIOD
  - -- SCOPING COMMENTS NOT INCLUDED
  - -- POSTPONE PENDING RELATED NEPA ACTIVITIES
  - -- REQUESTED ADDITIONAL PUBLIC HEARINGS
  - -- INAPPROPRIATE SCOPE
  - -- INSTALLATIONS NOT EQUALLY REPRESENTED
  - -- INSUFFICIENT STATEMENTS OF PURPOSE AND NEED
- o PROVIDES OPTIONS FOR TESTING ALTERNATIVE FUELS
- o REQUESTED CLARIFICATION RELATIVE TO INDEMNIFICATION

DEIS PUBLIC HEARING COMMENTS (cont'd)

- QUESTIONED RELATIONSHIP TO PREVIOUS CLASSIFIED DOCUMENTS
- IMPACTS ON YUCCA MOUNTAIN NOT ANALYZED
- POTENTIAL IMPACTS ON EMPLOYMENT NOT ANALYZED
- ALTERNATIVES:
  - PERSONNEL REQUIREMENTS UNDERESTIMATED
  - ALL SUITABLE SITES HAVE NOT BEEN INCLUDED
  - TRANSPORTATION ISSUES INADEQUATELY ADDRESSED
  - METEOROLOGICAL PREREQUISITES NOT SPECIFIC
  - NON-NUCLEAR ALTERNATIVES NOT CONSIDERED

DEIS PUBLIC HEARING COMMENTS (cont'd)

- HAZARDOUS MATERIALS:
  - HIGH LEVEL WASTE STREAM NOT IDENTIFIED
  - TRU WASTE CERTIFICATION MISREPRESENTED
  - WASTE REDUCTION TECHNIQUES NOT ADDRESSED
  - WIPP CAPABILITIES OVER ESTIMATED
  - SOME DISPOSAL METHODS ARE ILLEGAL
  - LIQUID WASTE STREAMS NOT ANALYZED
  - RCRA-LISTED WASTE NOT SPECIFIED
  - LIMITS OF INEL RCRA-B EXCEEDED
  - LDR WASTES NOT IDENTIFIED
- POTENTIAL GREENHOUSE EFFECT NOT ANALYZED

DEIS PUBLIC HEARING COMMENTS (cont'd)

- o PROPOSAL INSENSITIVE TO DESERT ECOLOGY
- o NATIVE AMERICAN ISSUES NOT ADDRESSED
- o GEOLOGY AND SOILS:
  - -- SEISMIC ACTIVITIES UNDERESTIMATED
  - -- VOLCANIC ACTIVITIES UNDERESTIMATED
  - -- FLOODING POTENTIAL UNDERESTIMATED
- o WATER RESOURCES:
  - -- POTENTIAL DROUGHT EFFECTS NOT ANALYZED
  - -- GROUNDWATER UNCERTAINTIES NOT IDENTIFIED

DEIS PUBLIC HEARING COMMENTS (cont'd)

- o HEALTH AND SAFETY
  - -- ALARA PRINCIPLE NOT SUFFICIENTLY EXPLAINED
  - -- HEAT RELATED SAFETY ISSUES NOT IDENTIFED
  - -- ACCIDENT ANALYSES NOT BOUNDING
  - -- INSUFFICIENT DESIGN DETAIL - CONTAINMENT & CONTROL
  - -- INSUFFICIENT DESIGN DETAIL - ETS
  - -- CONTROL ROOM HABITABILITY NOT DEFINED
  - -- INAPPROPRIATE CORRELATIONS TO CHEST X-RAYS
  - -- FALLOUT CONSEQUENCES TO FLORA/FAUNA NOT DISCUSSED

DEIS PUBLIC HEARING COMMENTS (cont'd)

- o HEALTH AND SAFETY (cont'd)
  - -- HYDROGEN SAFETY ISSUES NOT ADEQUATELY ADDRESSED
  - -- HYDROGEN EMBRITTLEMENT ISSUES NOT CONSIDERED
  - -- CONSEQUENCES TO SOUTHERN UTAH NOT SPECIFIED
  - -- METHODS TO CALCULATE EXPOSURE NOT EXPLAINED
  - -- QUANTITY OF REACTOR CORES ON-SITE NOT SPECIFIED
  - -- REQUESTED A DISCUSSION OF MILK MONITORING PROGRAMS
  - -- CLARIFY HAZARDS OF LONG TERM LOW-LEVEL EXPOSURES

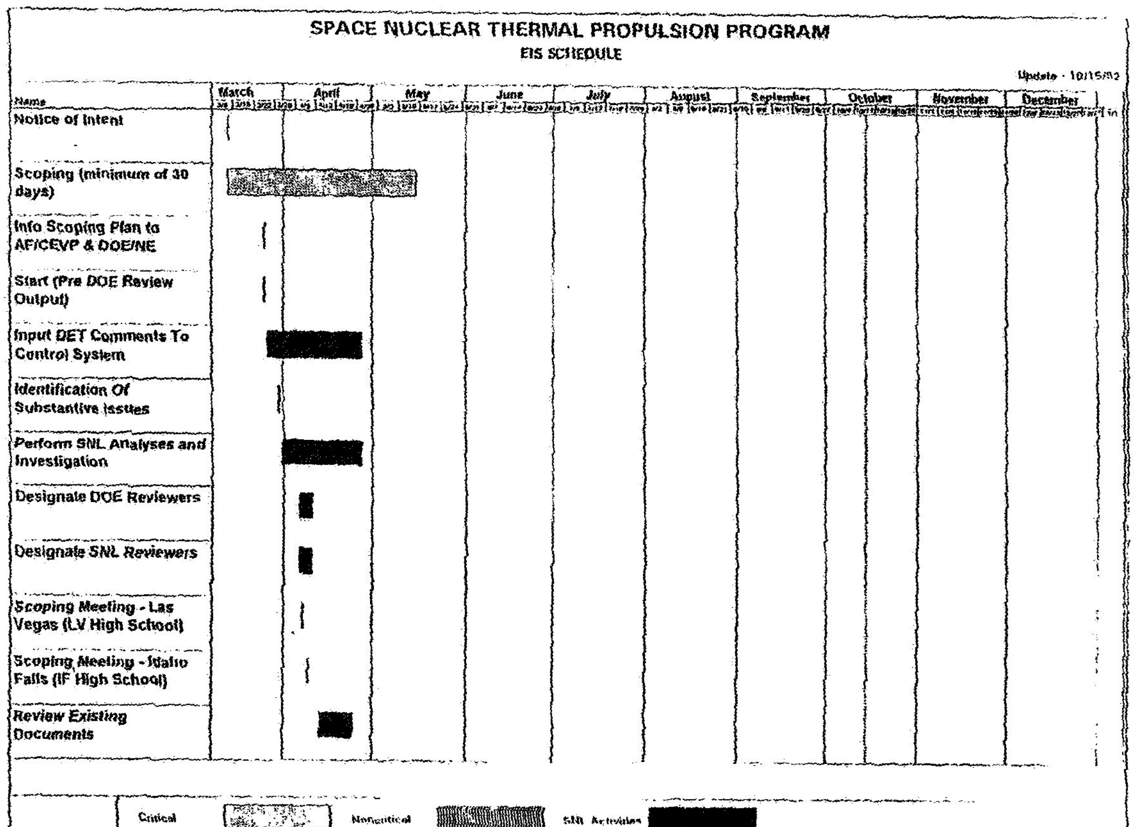
DEIS PUBLIC HEARING COMMENTS (cont'd)

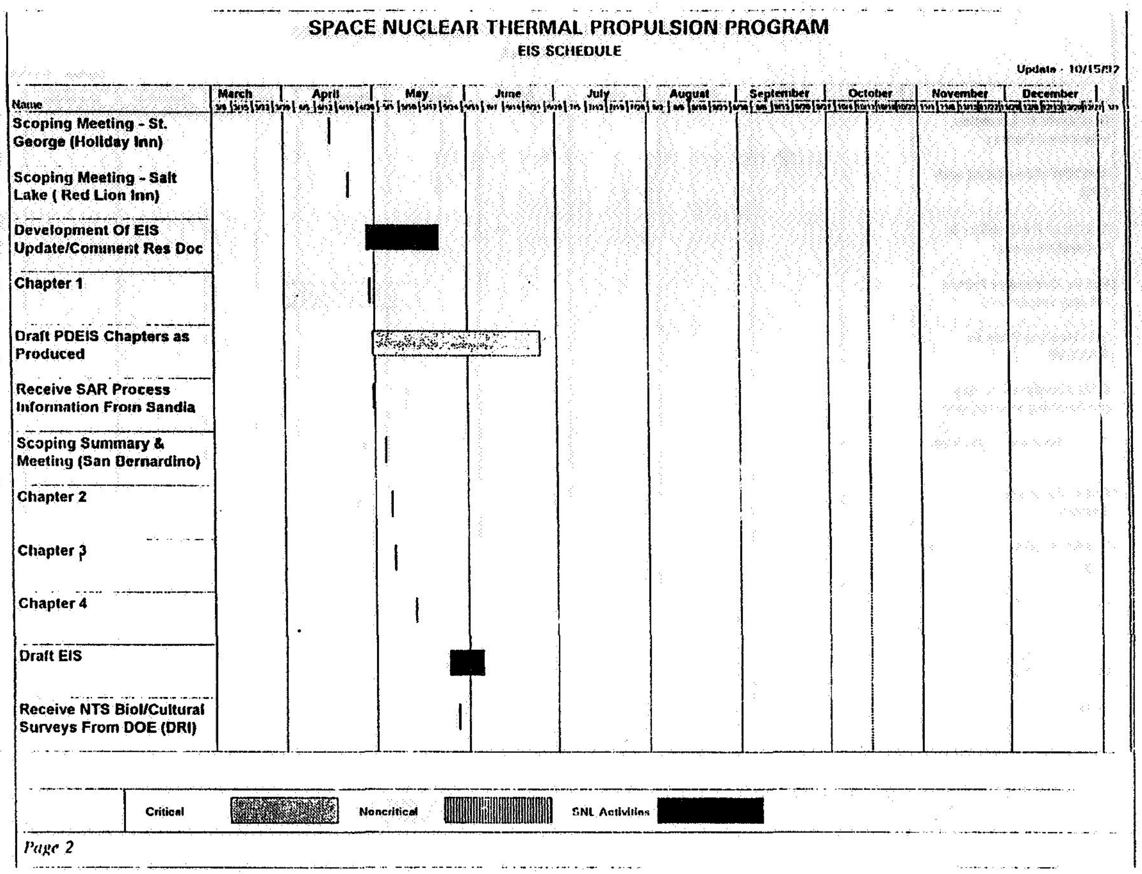
- o HEALTH AND SAFETY (cont'd):
  - -- NO-THRESHOLD-LEVEL CONCEPT NOT APPLIED
  - -- INVERSION LAYER EFFECTS ON Be RELEASES NOT DISCUSSED
  - -- BACKGROUND DOSES ARE TOO HIGH
  - -- LACK OF RAIL SYSTEMS NOT IDENTIFIED
  - -- PROGRAM BENEFITS NOT COMPARED TO RISKS
  - -- EFFECTS FROM HIGH VOLTAGE LINES NOT DISCUSSED
  - -- DOE SAR PROCESS INADEQUATE
  - -- 20% NESHAP REPRESENTS SIGNIFICANT INCREASES

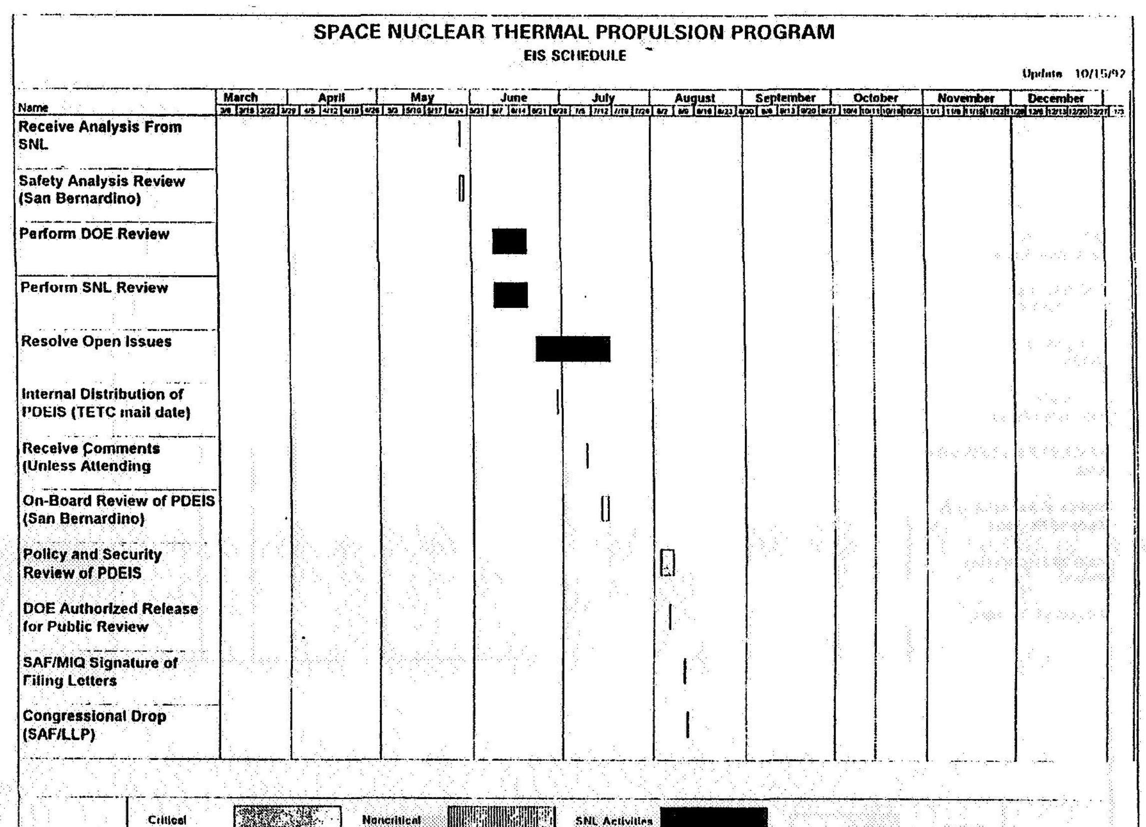
DEIS PUBLIC HEARING COMMENTS (cont'd)

o HEALTH AND SAFETY (cont'd)

- -- 170 mRem LIMIT NOT EXPLAINED
- -- CHARTS DO NOT SUPPORT DECREASING DOSE CLAIMS
- -- 1980 CENSUS DATA USED FOR INEL




SPACE NUCLEAR THERMAL PROPULSION PROGRAM
EIS SCHEDULE
Update · 10/15/92
Name
March
April
May
June
July
August
September
October
November
December
Scoping Meeting - St. George (Holiday Inn)
Scoping Meeting - Salt Lake (Red Lion Inn)
Development Of EIS Update/Comment Res Doc
Chapter 1
Draft PDEIS Chapters as Produced
Receive SAR Process Information From Sandia
Scoping Summary & Meeting (San Bernardino)
Chapter 2
Chapter 3
Chapter 4
Draft EIS
Receive NTS Biol/Cultural Surveys From DOE (DRI)
Critical
Noncritical
SNL Activities
Page 2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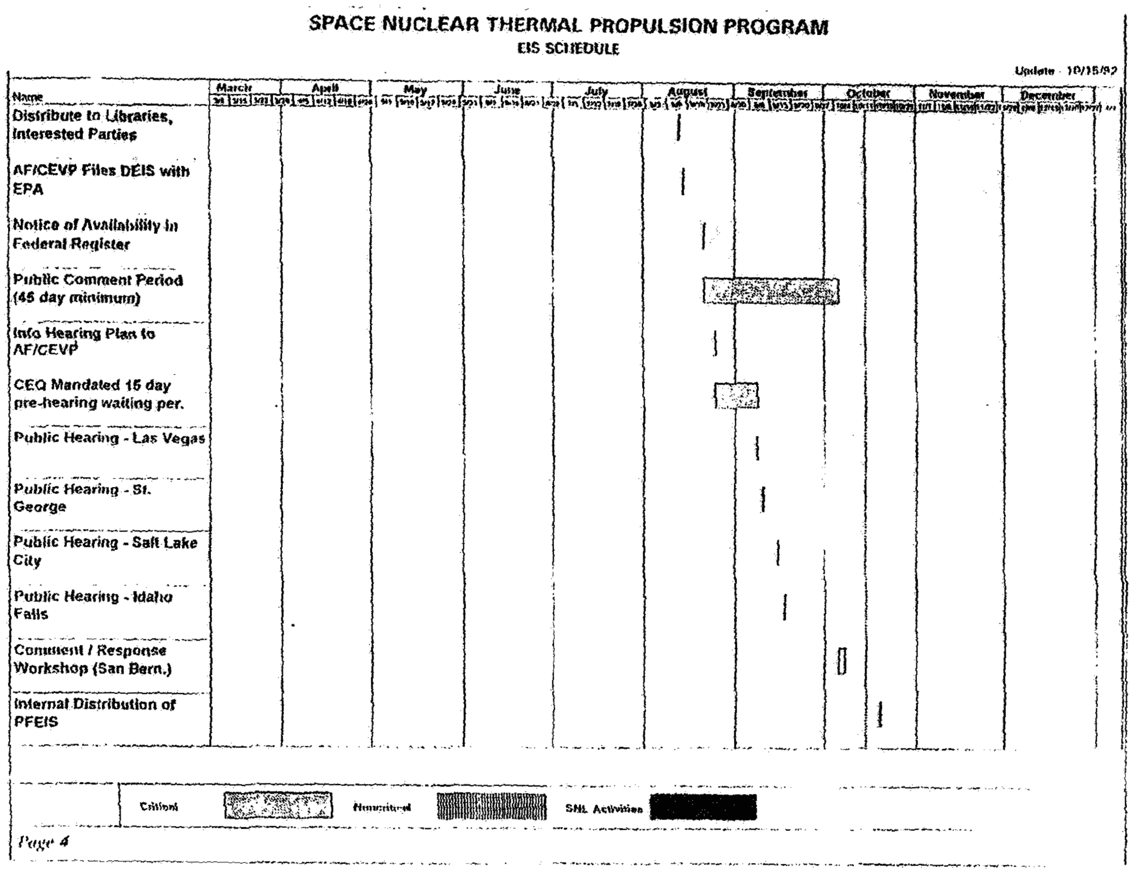
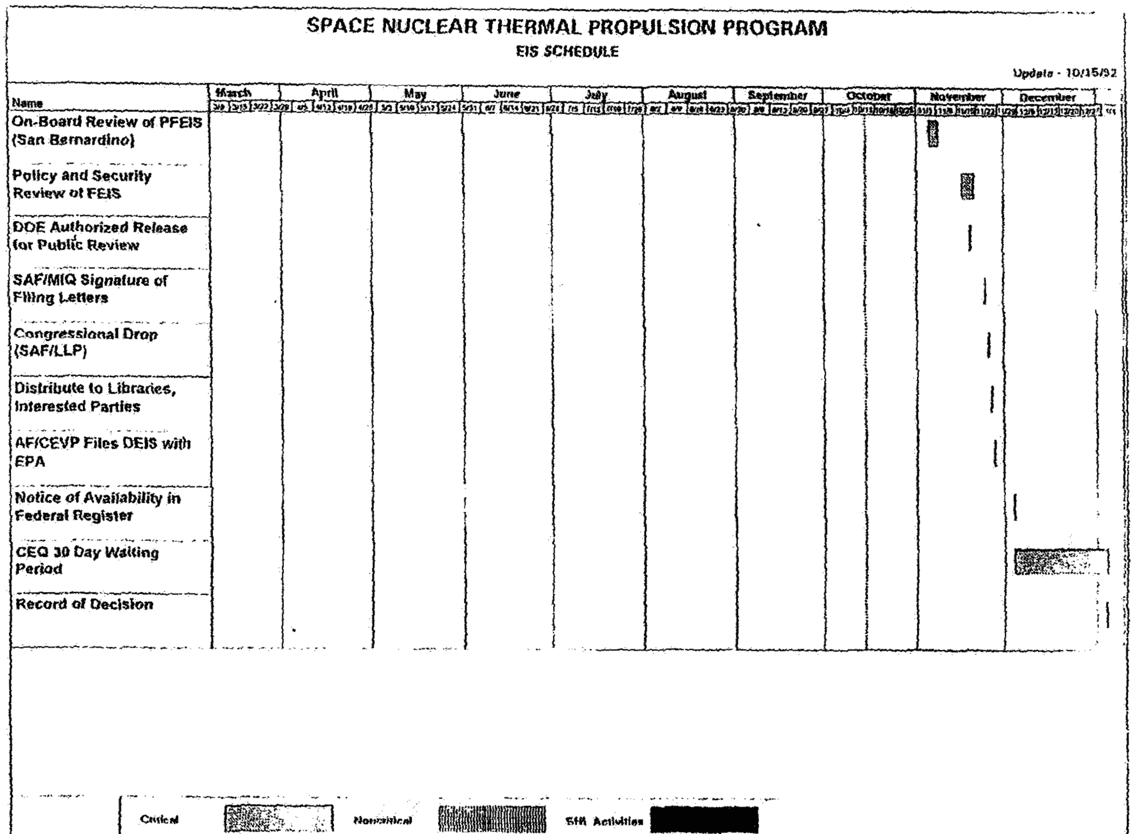
EIS SCHEDULE
Update 10/15/92
Name
March
April
May
June
July
August
September
October
November
December
Receive Analysis From SNL
Safety Analysis Review (San Bernardino)
Perform DOE Review
Perform SNL Review
Resolve Open Issues
Internal Distribution of PDEIS (TETC mail date)
Receive Comments (Unless Attending
On-Board Review of PDEIS (San Bernardino)
Policy and Security Review of PDEIS
DOE Authorized Release for Public Review
SAF/MIQ Signature of Filing Letters
Congressional Drop (SAF/LLP)
Critical
Noncritical
SNL Activities


SPACE NUCLEAR THERMAL PROPULSION PROGRAM
EIS SCHEDULE
Update - 10/15/92
Name
March
April
May
June
July
August
September
October
November
December
Distribute to Libraries, Interested Parties
AF/CEVP Files DEIS with EPA
Notice of Availability in Federal Register
Public Comment Period (45 day minimum)
Info Hearing Plan to AF/CEVP
CEQ Mandated 15 day pre-hearing waiting per.
Public Hearing - Las Vegas
Public Hearing - St. George
Public Hearing - Salt Lake City
Public Hearing - Idaho Falls
Comment / Response Workshop (San Bern.)
Internal Distribution of PFEIS
Critical
Noncritical
SNL Activities
Page 4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EIS SCHEDULE
Update - 10/15/92
Name
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September
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On-Board Review of PFEIS (San Bernardino)
Policy and Security Review of FEIS
DOE Authorized Release for Public Review
SAF/MIQ Signature of Filing Letters
Congressional Drop (SAF/LLP)
Distribute to Libraries, Interested Parties
AF/CEVP Files DEIS with EPA
Notice of Availability in Federal Register
CEQ 30 Day Waiting Period
Record of Decision
Critical
Noncritical

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13. ABSTRACT ***(Maximum 200 words)***

The Nuclear Propulsion Technical Interchange Meeting (NP-TIM-92) was held at NASA Lewis Research Center's Plum Brook Station in Sandusky, Ohio on October 20-23, 1992. Over 200 people attended the meeting from government, Department of Energy's national laboratories, industry, and academia. The meeting was sponsored and hosted by the Nuclear Propulsion Office at the NASA Lewis Research Center. The purpose of the meeting was to review the work performed in fiscal year 1992 in the areas of nuclear thermal and nuclear electric propulsion technology development. These proceedings are an accumulation of the presentations provided at the meeting along with annotations provided by the authors. The proceedings cover system concepts, technology development, and systems modeling for NTP and NEP. The test facilities required for the development of the nuclear propulsion systems are also discussed.

14. SUBJECT TERMS
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